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CONCEPTUAL STUDY OF A LOW COST TURBOJET ENGINE

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MARCH 1976

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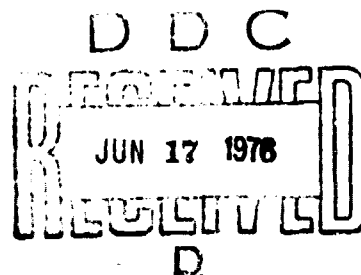
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CONCEPTUAL STUDY
OF A
LOW COST TURBOJET ENGINE
THESIS

GAE/AE/76M-2

Tommy J. Kent
Captain USAF



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a-b

GAE/AE/76M-2

CONCEPTUAL STUDY
OF A
LOW COST TURBOJET ENGINE

THESIS

Presented to the Faculty of the School of Engineering
of the Air Force Institute of Technology

Air University

In Partial Fulfillment of the
Requirements for the Degree of
Master of Science

By

Tommy J. Kent, B.S.

Captain USAF

Graduate Aero-Nautical Engineering

March 1976

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1-2

Preface

Defense officials believe that small inexpensive remotely-piloted aircraft can be built; however, the most serious problem at the moment is the lack of a reliable powerplant. This investigation of one solution to the problem was the first of a continuing study at AFIT. The study proved that an engine constructed from a supercharger can produce 60 lb or more of thrust and revealed many areas of interest for more detailed studies.

I would like to extend my sincerest thanks to Dr. William C. Elrod and Dr. Harold E. Wright of the Air Force Institute of Technology for their invaluable assistance throughout this investigation. I also acknowledge my debt to Mr. David Wilkinson of the Air Force Aero Propulsion Laboratory for practical ideas and suggestions from his experience with many types of engines. Also, I would like to thank Mr. Parks, Mr. Baker, and Mr. Flahive of the AFIT Labs for their assistance. Additionally, a special note of thanks is due Mr. Wolfe, Mr. Brohas, Mr. Murry, Mr. Grube, and Mr. Shortt, all from the AFIT machine shop. They built the parts for the machine and contributed many helpful suggestions.

Lastly, I submit that the patience, and understanding of my wife were key ingredients in the completion of this work.

TOMMY J. KENT

TABLE OF CONTENTS

	Page
Preface	ii
List of Figures	v
List of Tables	vii
List of Symbols	viii
Abstract	x
I. Introduction	1
Background	1
Objective	1
Scope	2
II. Description of Apparatus	5
Turbocharger-Engines	5
Test Equipment	10
III. Performance Analysis	16
Parameter Investigation	16
Performance	18
Results Applied to Experimental Investigation	21
IV. Experimental Results	26
Engine Stability	26
Turbine Housings	27
Air-flow Delivery	27
Combustion Chamber and Exit Nozzle	27
Lubrication System	35
Discussion of Performance Trends	36
V. Conclusions and Recommendations	39
Conclusions	39
Recommendations	39
Bibliography	42
Appendix A: Test Data	43
Appendix B: Performance Analysis	46

	Page
Appendix C: Instrumentation System	58
Appendix D: Detailed Calculations and Data Reduction	70
Vita	72

LIST OF FIGURES

Figure		Page
1	J1 Engine	3
2	J2 Engine	4
3	Configuration and Major Dimensions of the Turbo-charger Engines Tested	6
4	Vaneless Turbine Housing	8
5	Schematic of the Test Equipment	11
6	Fuel System	12
7	Oil System	13
8	Instrumentation System	14
9	Relative Importance of Performance Parameters . . .	17
10	Typical Compressor Map for a Turbocharger	19
11	Predicted Performance for the J1 and J2 Engine . .	20
12	Major Variables from the Thrust Equation	22
13	Effect of η_t on Thrust and P_{rn}	23
14	Engine Performance: Augmented and Dry Thrust in Corrected Variables	24
15	Turbine Housing Performance	28
16	Air-flow for the J1 Engine	29
17	Air-flow for the J2 Engine	30
18	Increased Air-flow Causes Higher Thrust	31
19	Combustion Chamber Performance	32
20	Exit Nozzle Performance from Direct Thrust Measurement	34
21	Performance Trends	38
22	Direct Thrust Measurement Data Sample	45
23	Relative Importance of Performance Parameters (Thrust Orientation)	48

24	Relative Importance of Performance Parameters (SFC Orientation)	49
25	Theoretical J1 Engine Performance	51
26	Theoretical J2 Engine Performance	52
27	Theoretical J2 Engine Thrust with Major Parameters .	53
28	Comparison of Theoretical Engine Performance	55
29	Specific Augmented and Dry Performance	56
30	J1 Engine Bellmouth Characteristic Curve	62
31	J2 Engine Bellmouth Characteristic Curve	63
32	Fuel Flow Meter Calibration Curve	64
33	Fuel Nozzle Performance	65

LIST OF TABLES

Table	Page
I. Major Dimensions of the Turbocharger-Engines Tested	6
II. Weights of Components for the Turbocharger-Engines	6
III. Cross-sectional Area and Mach Number	7
IV. Turbine Housings	8
V. Test Data	44
VI. Turbocharger-Engine Components and Performance Parameters	46

LIST OF SYMBOLS

A	Cross-sectional area of a duct
A/B	Afterburner (one form of thrust augmentation)
A/R	Turbine nozzle throat area/Radius from the centerline of the turbocharger to the centroid of the turbine nozzle throat area (used to differentiate various turbine housings)
C_p	Specific heat at constant pressure (BTU/lb _m R)
D	Outside diameter of the supercharger at its largest point (in.)
DRY	No thrust augmentation
FF	Fuel flow rate (gal/hr)
F_n	Thrust (lb _f)
f	Fuel to air ratio (lb _m fuel/lb _m air)
L	Length of the supercharger (in.)
L_b	Length of the combustion chamber liner (in.)
M	Mach Number
N	Speed of rotation (RPM)
P	Static pressure (lb _f /in. ² or in.Hg gage)
P_f	Static pressure of the fuel delivered to the fuel nozzle (lb _f /in. ²)
P_{fn}	Static pressure on the discharge side of the fuel nozzle (lb _f /in. ²)
P_{∞}	Ambient pressure (lb _f /in. ² or in.Hg absolute)
P_o	Total pressure (lb _f /in. ²)
P_{rc}	Compressor Pressure ratio (P_{o3}/P_{o2})
P_{rn}	Nozzle Pressure ratio (P_{o5}/P_{∞})
P_{rt}	Turbine Pressure ratio (P_{o4}/P_{o5})
ΔP_{ob}	Total pressure loss in the Combustion Chamber ($P_{o2}-P_{o3}/P_{o2}$)

ΔP_f Static Pressure drop across fuel nozzle ($\text{lb}_f/\text{in.}^2$)
 Q Heating value of the fuel ($\frac{\text{BTU}}{\text{lb}_m}$)
 R Gas constant ($\frac{\text{ft} \cdot \text{lb}_f}{\text{lb}_m \text{R}}$)
 R Temperature (Rankine)
 RPM Revolutions per minute
 SFC Specific fuel consumption (lb_m fuel per hr/ lb_f thrust)
 T Static temperature (F or R)
 T_o Total temperature (F or R)
 W_a Weight flow of air (lb_m/sec)
 W_f Weight flow of fuel (lb_m/sec)
 γ Ratio of specific heats
 δ Ratio of pressures ($P_{\infty}/29.92 \text{ in. Hg}$)
 η_b Combustion chamber efficiency
 η_c Compressor efficiency
 η_n Nozzle efficiency
 η_t Turbine efficiency
 θ Ratio of temperature ($T_{\infty}/519 \text{ R}$)

Subscripts

∞ Ambient condition
 2 Compressor entrance
 3 Between compressor and combustion chamber
 B In combustion chamber
 4 Between combustion chamber and turbine
 5 Between turbine and nozzle
 6 Nozzle exit
 cor Corrected to standard conditions

ABSTRACT

One candidate for an inexpensive engine is a turbojet constructed by adding a combustion chamber and nozzle to a commercial turbocharger for a reciprocating engine. Two turbocharger-engines were produced in this way capable of handling approximately .6 lb/sec and 1.5 lb/sec airflow. A performance analysis using Thermodynamic Cycle Analysis Techniques was done to predict the thrust that could be generated by these engines. The maximum thrust predicted was 27 lb and 67 lb dry; with an afterburner, 36 lb and 96 lb was expected. A combustion chamber and nozzle were added to the turbochargers and both were mounted on a test stand and successfully operated.

Thrust was increased by various improvements to 60 lb. The performance parameters of possible interest were studied with particular emphasis placed on thrust, weight flow of the air, specific fuel consumption, compressor pressure ratio, and temperature of the gases at the compressor, turbine, and nozzle.

CONCEPTUAL STUDY OF A LOW COST TURBOJET ENGINE

I. Introduction

Background

The Air Force Aero Propulsion Laboratory is studying possible propulsion devices for small remotely piloted vehicles (RPVs). Modification of a turbocharger (turbo-supercharger) for a reciprocating engine to produce a low-cost turbojet engine is one approach being considered.

Present turbojet engines are in a higher thrust range and cost much more than this engine. Typical current low thrust engines cost approximately \$73 per lb of thrust, but a turbocharger-engine (an engine constructed from a turbocharger) will cost approximately \$22 per lb of thrust. The possibility of large cost savings exists if this low-cost engine's performance is adequate. Also, this type of engine will be ideal for the application described by Klass (Ref 1).

Defense officials believe that mini-RPVs, ranging in size from 60-120 lb, can be built in quantities of several thousand to sell for \$10,000 - \$20,000 each, including avionics payload. Some are designed as expendables for one-way missions.

The most serious problem at the moment is the lack of long-mission reliability of small piston engines, originally designed for low-cost civil applications such as "Go-Karts", chain saws and model aircraft.

Objective

The primary objective of the investigation was to develop a turbojet engine by adding an inlet bellmouth, combustion chamber, and nozzle to a turbocharger designed for use on a

reciprocating engine. Secondary objectives were to determine the primary performance parameters and operate such an engine as a turbojet to obtain performance data.

Scope

The analytic investigation was limited to use of Carpet (a thermodynamic cycle analysis computer program, Ref 5). No attempt was made to use computerized compressor-turbine matching techniques.

The experimental investigation was limited to two different size engines: the J1 Engine, constructed from a Rajay 307 E Turbocharger and the J2 Engine, the larger of the two, constructed from an AiResearch T18A E Turbocharger. The investigation was also limited to those turbine housings available from the manufacturers of these turbochargers.

No attempt was made to limit or reduce the weight of engine components and the lubrication system was improved only to the point where the engines could be operated long enough to obtain performance data. No attempt was made in this investigation to simulate a flyable engine. Modifications were limited to the addition of a combustion chamber for each engine (two combustion chambers) and two types of nozzle configuration (one with adjustable exit area, the other with the exit area fixed).

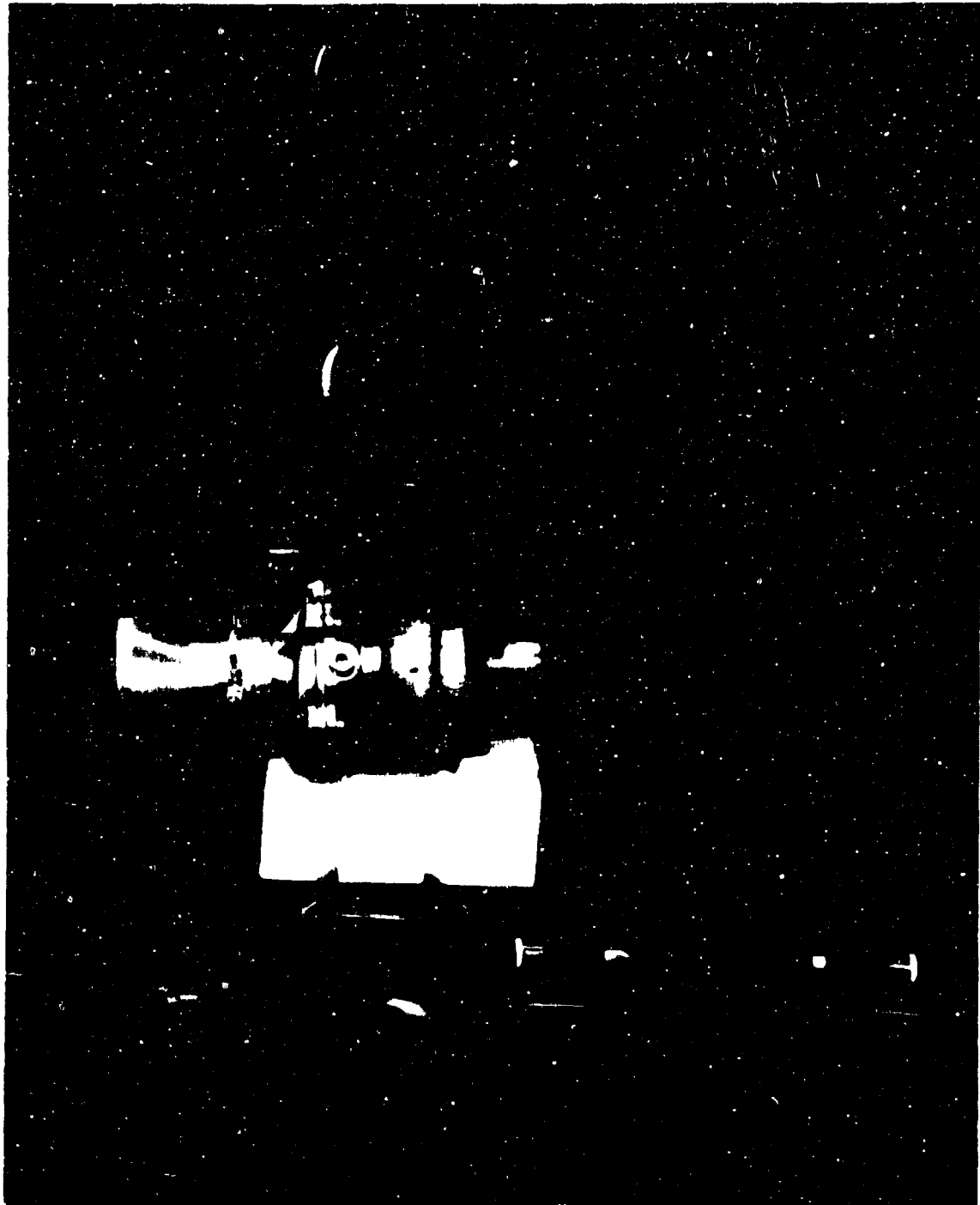


Fig 1. J1 Engine

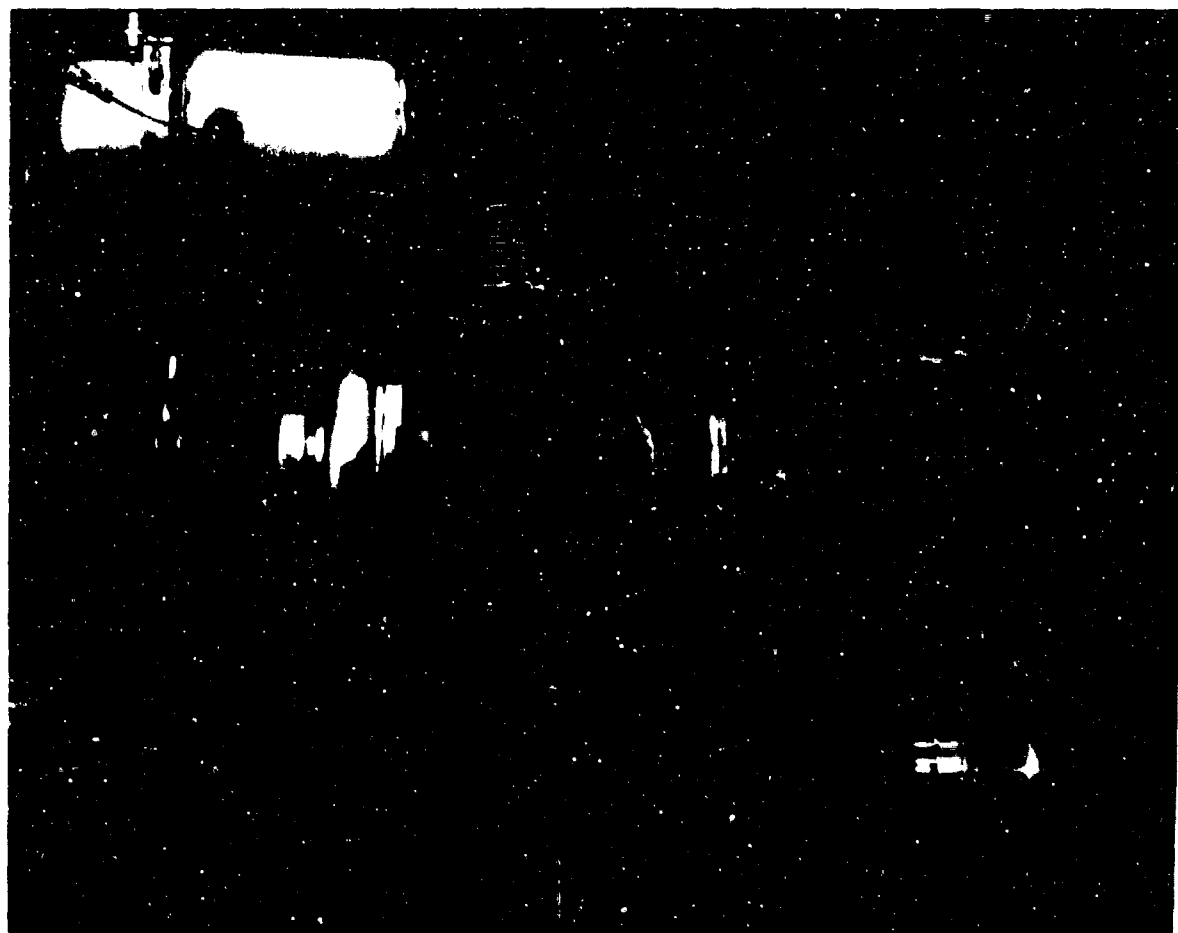


Fig 2. J2 Engine

II. Description of Apparatus

The turbocharger-engines were constructed by adding the necessary components to the turbochargers as they were received from the manufacturer. A combustion chamber and nozzle were the two major additions. The combustion chamber was a major design effort (see Greene Ref 2) and was the most complicated addition made. Considerable additional equipment was necessary to supply the machines with fuel and oil, and to measure the parameters involved.

Turbocharger-Engines

Pictures of the two engines tested are shown in Figs 1 and 2. The general layout and major components are shown schematically in Fig 3; however, the size of the machine is not indicated by this schematic. The only major differences in the machines were size and the combustion chamber used on each. Tables I and II give the major dimensions and weight of components for each engine. Note the high relative weight of the turbine housing for each machine. This part is designed to contain the turbine wheel fragments if it disintegrates at high speed. The relatively crude design of the combustion chamber for the J1 Engine was dictated by ease of construction and availability of materials and parts. The J2 Engine combustion chamber is a more sophisticated design adapted from a MA-1A ground support unit.

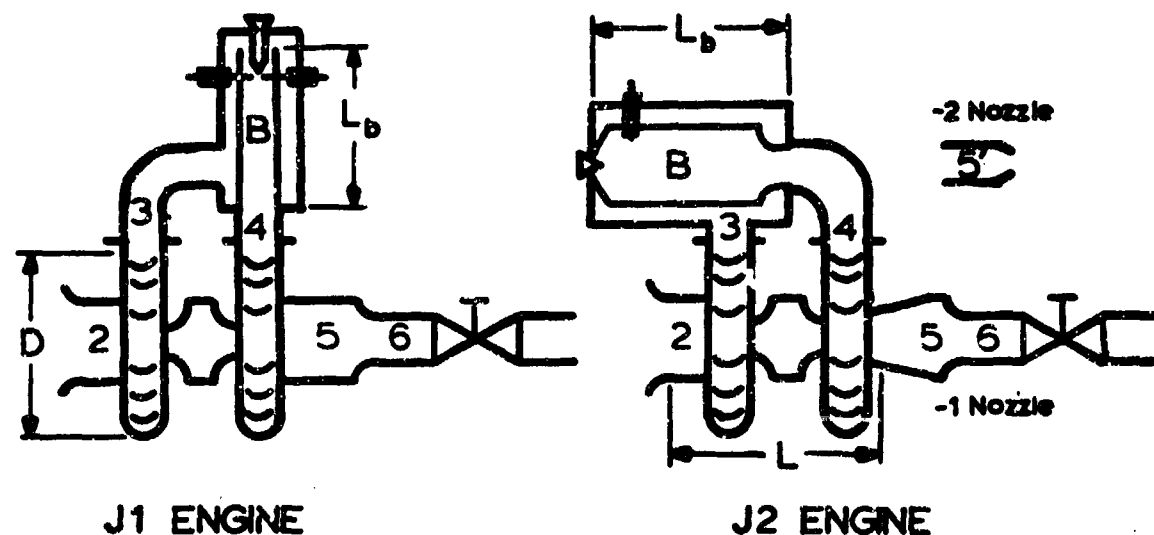


Fig 3. Configuration and Major Dimensions of the Turbocharger-Engines Tested

Table I

Major Dimensions of the Turbocharger Engines

Note: Length does not include bellmouth or exit duct

Dimension	J1 Engine	J2 Engine
	inches	inches
D	7.2	11.2
L	6.87	10.87
L _b	12.00	13.10

Table II

Weight of Components for the Turbocharger Engines

Note: Weight is in lb

Component	J1 Engine	J2 Engine
Compressor Housing	2.0	3.4
Compressor, Turbine Wheel and Bearing Assembly	4.0	13.8
Turbine Housing	8.5	30.5
Combustion Chamber	16.5	13.6
Total	31.0	66.3

Table III

Cross-sectional Area And Mach Number

Mach number calculated for maximum W_a with P_o from Carpet (see Appendix D).

Note: a = entrance to duct indicated on Fig 3; b = exit of duct; B = Combustion chamber areas; Ba = total area inside plenum including liner; Bb = annulus area between outer wall and liner; Bc = liner area; Bd = exit area; 5' = Fixed area nozzle.

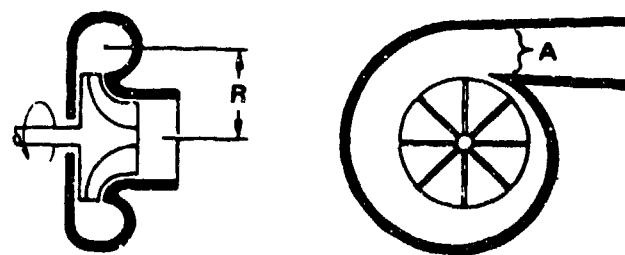
Location	Shape	J1 Engine		J2 Engine	
		Area	M	Area	M
2	a	5.412	0.21	12.927	0.19
	b	3.80	0.31	8.55	0.29
3	a	2.99	0.17	7.07	0.11
	b	3.05	0.16	8.30	0.09
B	a	9.90	—	41.28	—
	b	4.99	0.10	19.63	0.04
	c	4.43	0.21	20.63	0.07
	d	4.43	0.21	16.62	0.08
4	a	5.31	0.17	8.76	0.15
	b	4.13 ^a	0.23	7.66 ^b	0.18
5	a	4.91	0.39	11.64	0.29
	b	5.94	0.31	21.65	0.15
6	-	3.05	0.61	7.451	0.49
5'	a	—	—	11.64	0.29
	b	—	—	5.940	.718

^a2.31 x 1.88 with 0.50 corner radius

^b3.42 x 2.38 with 0.75 corner radius

Table III is a comparison of the relative size and Mach number in flow passages of the two engines. See Appendix D for calculation details.

Each engine had a number of different size vaneless turbine housings. The casting itself serves the purpose of guide vanes for the turbine.



A = area of throat

R = distance from centroid
of this area to the
center of the turbine
wheel

Fig 4. Vaneless Turbine Housing (Ref 3)

The ratio A/R determines the amount of power extracted from the hot gas as it flows through the turbine. At a fixed W_a a large A/R will decrease the gas velocity and allow the gas to flow at a steep angle as it enters the turbine blades. The low velocity and steep entry angle cause the turbine wheel to rotate slower than it would with a small A/R . The turbine housings available for this investigation are listed in Table IV.

Table IV
Turbine Housings

J1 Engine		J2 Engine	
Housing	A/R	Housing	A/R
1	0.7	1	1.0
2	0.8	2	1.14
3	0.9	3	1.32
4	1.0	4	1.50
		5	1.70

Both of these engines use sleeve type bearings. The J1 Engine uses a semi-floating bearing, cast as one piece, which forms both radial load and thrust bearings. The J2 Engine has three separate floating bearings. All bearing surfaces in both machines are supplied with pressurized oil by channels drilled directly to the supply port. See Ref 3 for a detailed discussion of the bearings and oil system.

Two types of nozzles (see Fig 3) were used on these engines. For most of the investigation a variable area nozzle was used so the back pressure could be adjusted. The valve shown is a simple brass gate valve (water was injected upstream for cooling). This nozzle will be referred to by dash number 1. The dash 2 fixed area nozzle was used for the last few data points.

It will be convenient to refer to the engines by a numbering system specifying the turbocharger, housing, and nozzle used. The "J" in front of the word engine signifies a turbo-jet engine. The number following the "J" identifies the turbocharger used to construct the engine. The "A" or "B" after this number specifies which of the two turbochargers of each type was being used. The first dash number refers to

the turbine housing and the second to the nozzle. For example, J2A-4-2 Engine is identified as one constructed from the larger Turbocharger by a "2" after the "J". "A" signifies it was constructed from the first of the two turbochargers available. The -4 refers to turbine housing #4 (see Table IV) and the -2 refers to the constant area exit nozzle described above.

Test Equipment

The rocket engine test facilities of the Aero-Mechanical Engineering Department (Ref 4) were used to test these engines. A schematic of the test equipment is shown in Fig 5. Five major support systems were necessary:

1. Fuel system
2. Ignition system
3. Oil system
4. Starter
5. Instrumentation system

Fuel was supplied to the fuel nozzle in the combustion chamber from the pressurized tank shown in Fig 6. Flow rate was controlled by the needle valve and nozzle size.

The fuel/air mixture was ignited by a 10,000 volt furnace ignitor system. Electrodes mounted on the outer wall of the combustion chamber were used for the J1 Engine; however, a simple spark plug type ignitor was used for the J2 Engine.

The oil system shown in Fig 7 was used to supply oil to the bearings for cooling and lubrication. The oil was pumped by the gear pump through a pressure regulating valve and a

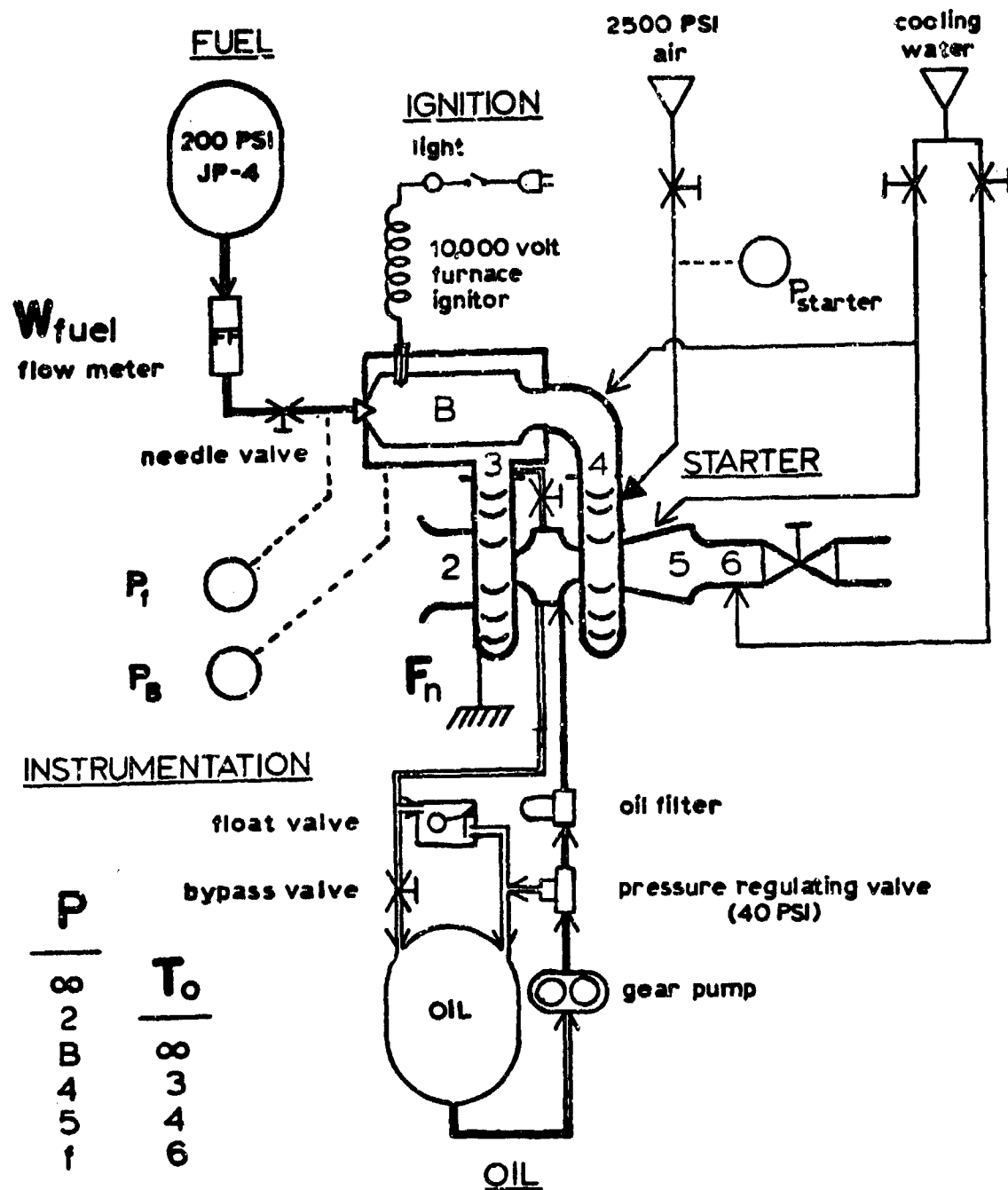


Fig 5. Schematic of the Test Equipment

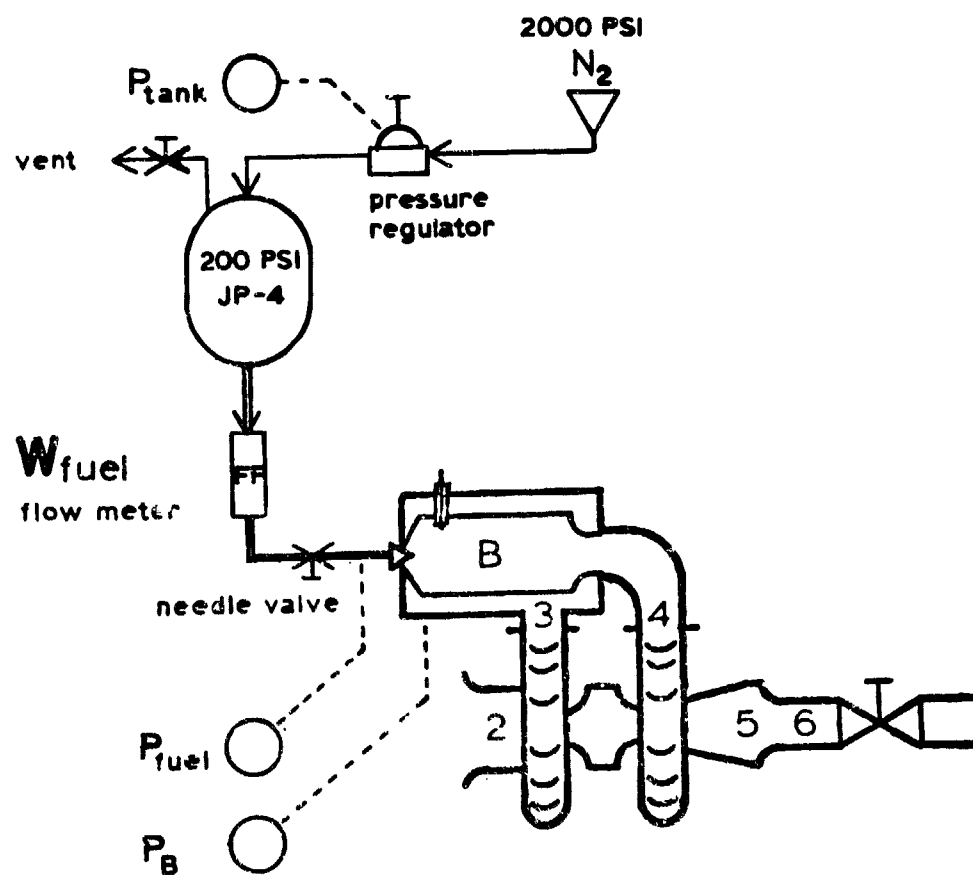


Fig 6: Fuel System

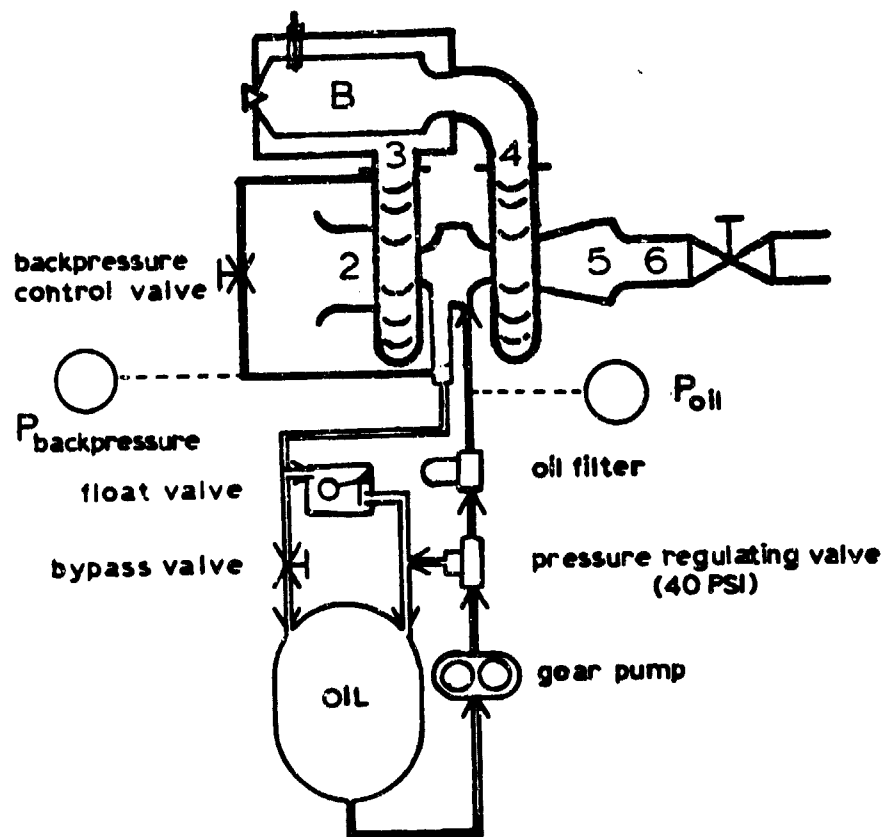


Fig 7: Oil System

filter to provide an adequate supply of clean oil to the bearings. By use of the float valve, the oil exit chamber and return lines could be pressurized by the compressor.

The starter was a simple 1/4 inch line welded into the turbine housing. It allowed a high velocity air jet to impinge on the turbine blades.

See Appendix C for details of the instrumentation system used to measure the parameters shown on Fig 8 and the left and lower part of Fig 5.

III. Performance Analysis

The "Design Point Turbine Engine Performance Program" (Ref 5) developed by the Air Force Aero Propulsion Laboratory was used to conduct the performance analysis of the turbocharger engines that is described in Appendix B. Performance parameters were studied individually; then, the engine operation was simulated more closely by including data from compressor maps for each of the engines. The results of this analysis are useful for understanding the operation of the turbocharger-engine, but more importantly, the results give an indication of the performance to be expected and provide a basis for determining the course of the experimental investigation.

Parameter Investigation

The study of individual parameters revealed significant difference in their effect on thrust (F_n). The parameters are listed from left to right in order of importance on Fig 9. The graph shows the effect of changing a single parameter when the others are held constant at the reference value listed below each individual plot.

W_a has the most effect on thrust as shown by comparing the two different engines. When the effect of W_a is removed, by using the scale on the right (F_n/W_a) there is little difference in specific thrust for two engines.

T_{04} is the next parameter in order of importance. There is a large increase in F_n as the turbine temperature is increased. The exact maximum value of this temperature is

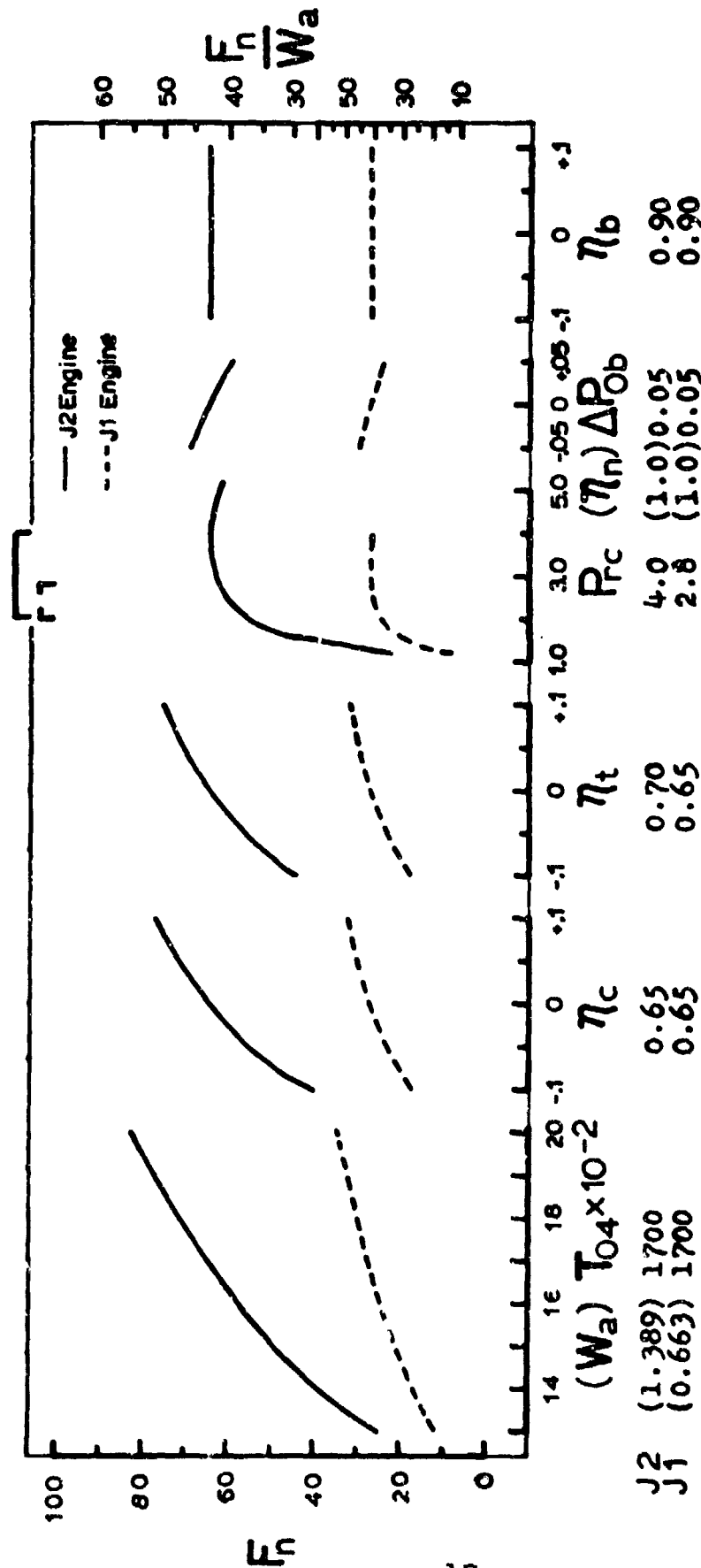


Fig 9: Relative Importance of Performance Parameters

limited by the turbine materials, and is yet to be determined.

η_c and η_t are also determined by the machinery and could not be controlled in this investigation.

A range of operation is shown for P_{rc} . A significant percent of the maximum thrust can be achieved over a wide range, provided P_{rc} is greater than 2.0, and all other parameters are constant. However, in normal turbojet operation, this parameter is a function of throttle position (W_f) and can be easily correlated with RPM by a compressor map (see Fig 10) which makes it a convenient reference. There is no advantage in pushing P_{rc} past the maximum thrust point because of the increase in stress on the machinery as RPM is increased.

Performance

An assumed operating line on the compressor map, positioned to operate the compressor at peak efficiency (see Fig 10) was used to analyze the performance of the engines over the complete operating range. Fig 11 shows the resulting predicted performance for both engines. The curves are very similar in shape; however, the J2 Engine allows operation at higher compressor pressure ratios (P_{rc}), higher air-flow rates (W_a), lower SFC, and a higher thrust (F_n) to be produced.

A better understanding of the parameter: that effect thrust can be obtained from the thrust equation:

$$F_n/W_a = (1+f) \sqrt{2 C_p T_{05} \eta_n \left[1 - \left(\frac{1}{P_{rn}} \right)^{\frac{\gamma-1}{\gamma}} \right]} \quad (1)$$

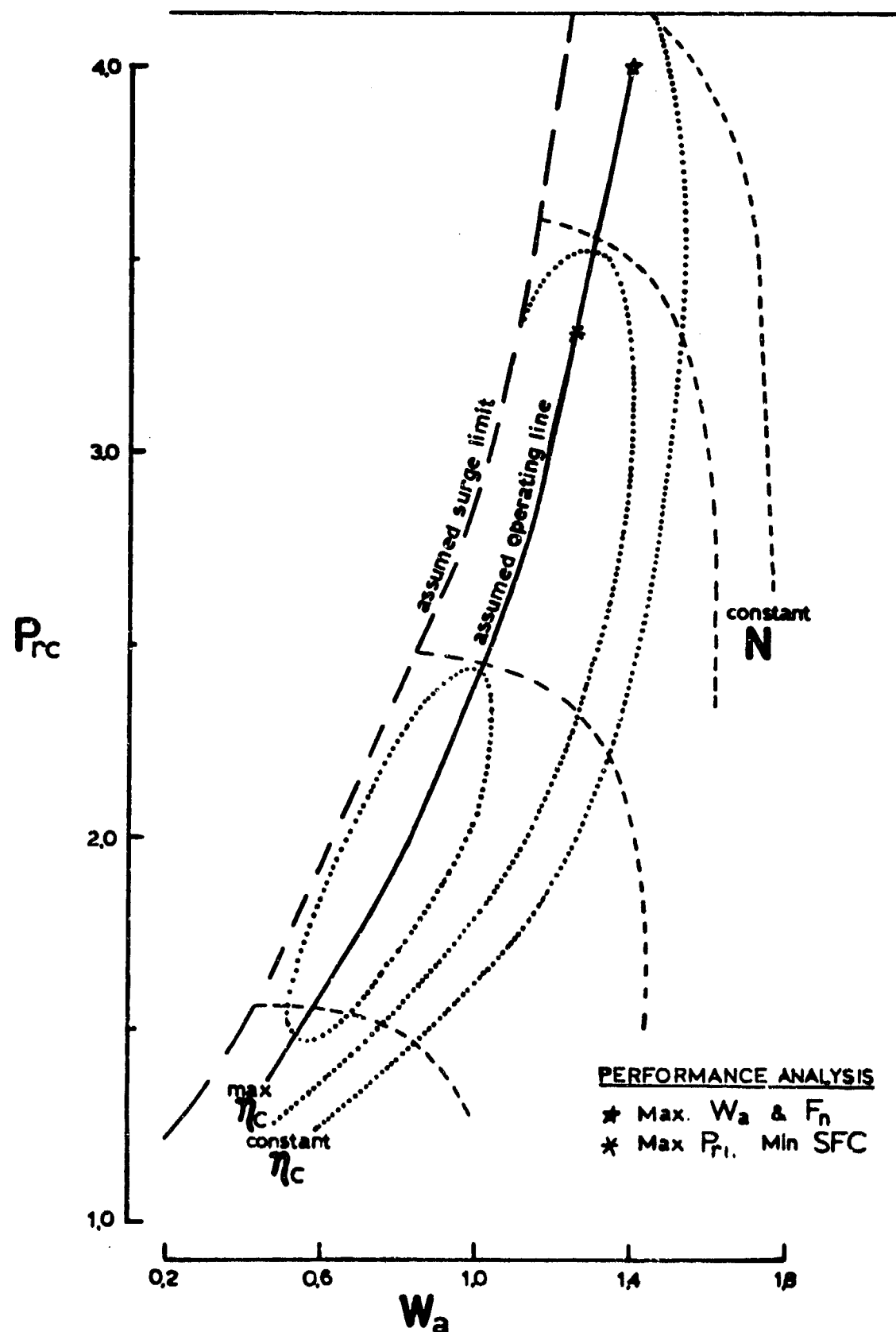


Fig 10: Typical Compressor Map for a Turbocharger

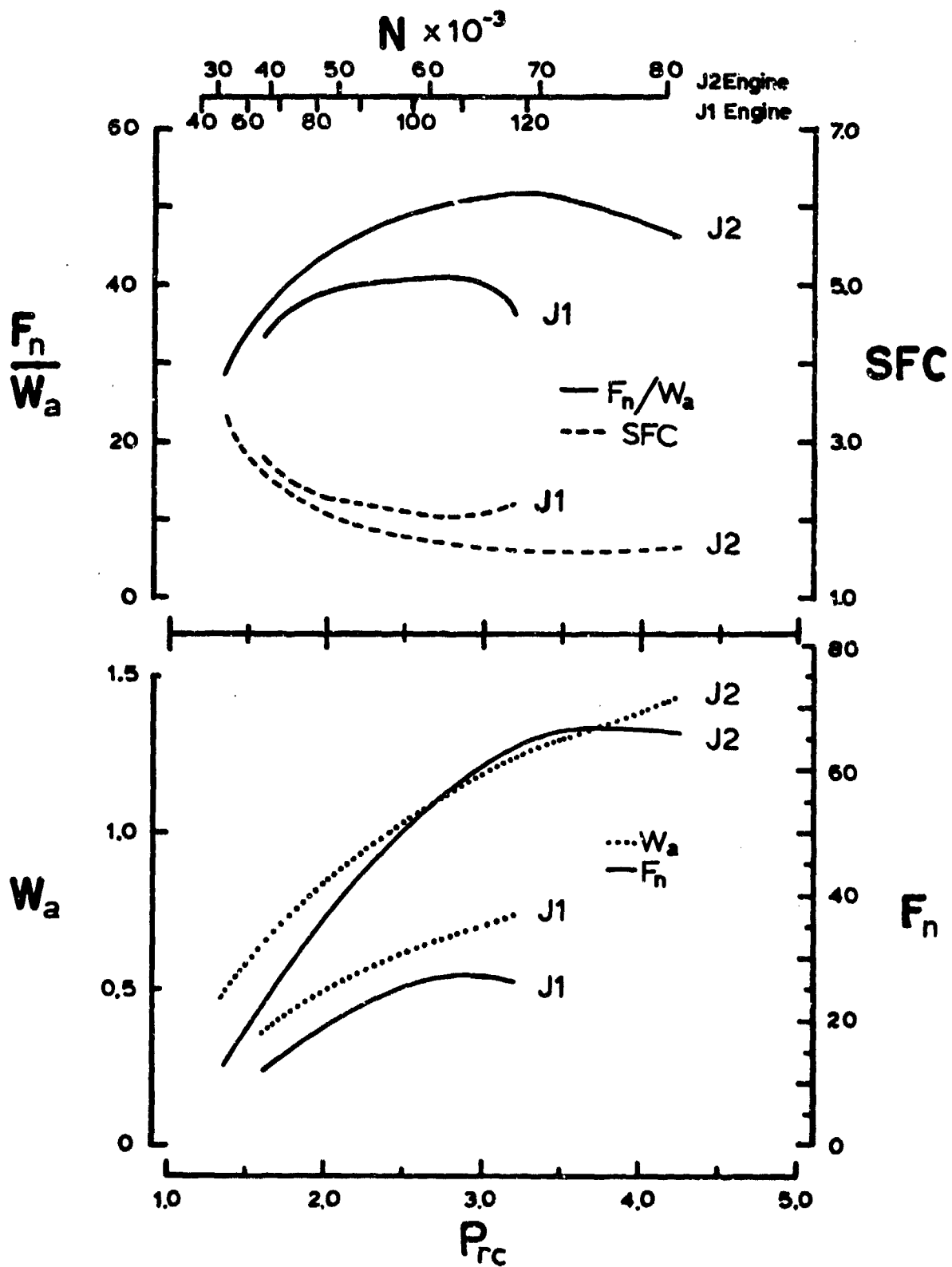


Fig 11: Predicted Performance for the J1 and J2 Engine

Two of the variables $f = .02$ and $\eta_n = 1.0$ may be assumed constant and C_p and γ depend on T_{05} . So F_n depends on three variables from this equation; P_{rn} , T_{05} and W_a . Fig 12 shows how these parameters interacted over the same range as those plotted in Fig 11; specific thrust (F_n/W_a) is common to both figures. Fig 12 shows thrust is a strong function of P_{rn} , T_{05} and W_a . However, Fig 13 shows turbine efficiency (η_t) has only a weak effect on thrust or P_{rn} . These figures show P_{rn} and F_n are good indicators of engine performance.

The predicted augmented thrust performance for a maximum afterburner operating temperature of 3500 F is compared to dry performance on Fig 14 in corrected variables. The maximum dry thrust for the J1 Engine can be increased by 33%. And the maximum for the J2 Engine by 43%. See Fig 28 in Appendix B for more detailed augmented thrust performance.

Results Applied To Experimental Investigation

The results of the performance analysis had a direct bearing on the experimental investigation. P_{rn} and F_n were used as indicators of engine performance. An adjustable back pressure nozzle was designed and various turbine housings evaluated by comparing performance data taken at the peak P_{rn} .

It was decided that all the turbine housing available from the manufacturer would be tested to obtain the compressor-turbine match that would allow maximum P_{rc} and P_{rn} . If $P_{rc} \geq 2.0$ could be achieved, significant thrust levels and performance data could be measured.

The parameter sensitivity analysis (see Appendix C) indicated that extreme instrumentation accuracy would be needed.

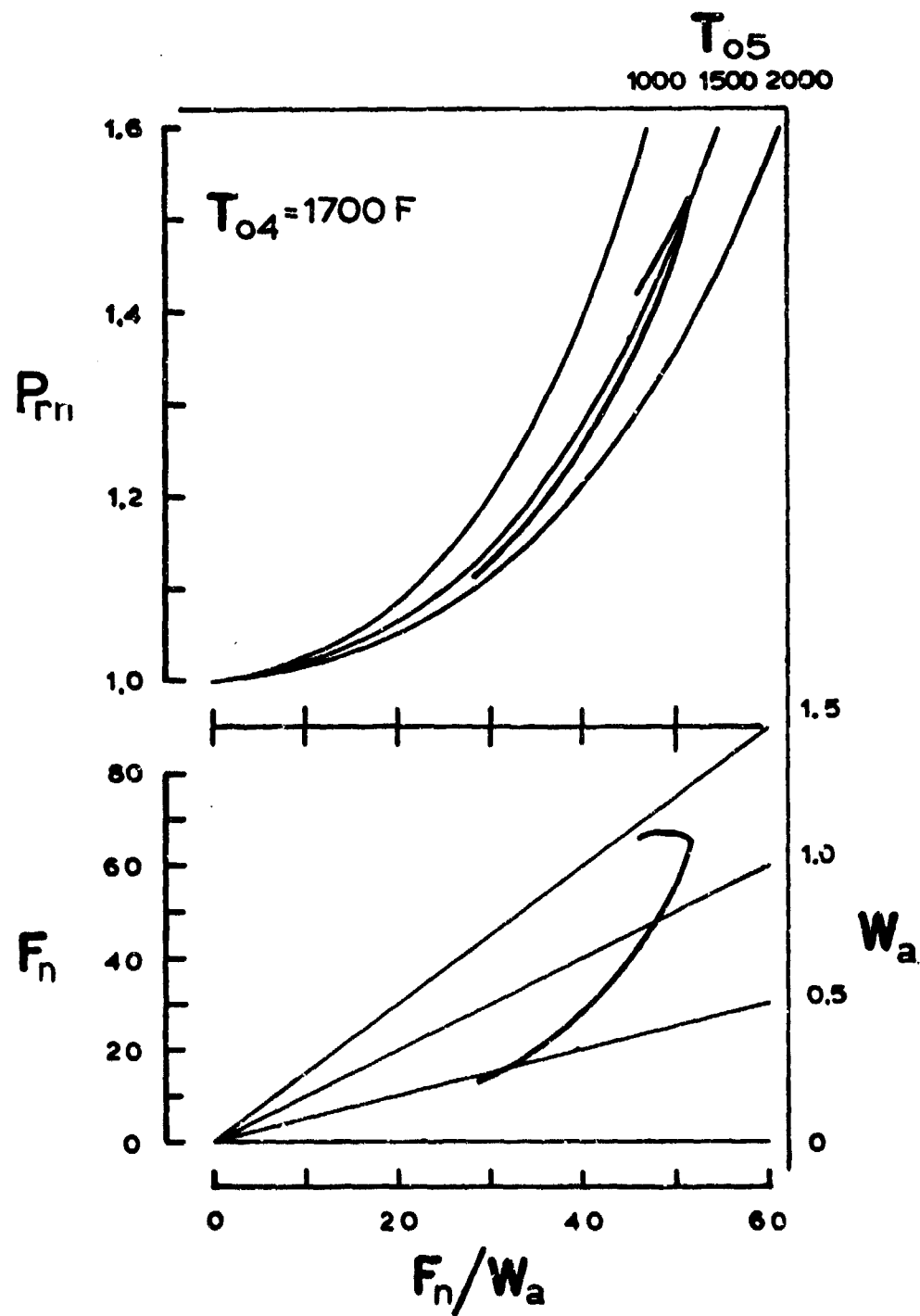


Fig 12: Major Variables from the Thrust Equation

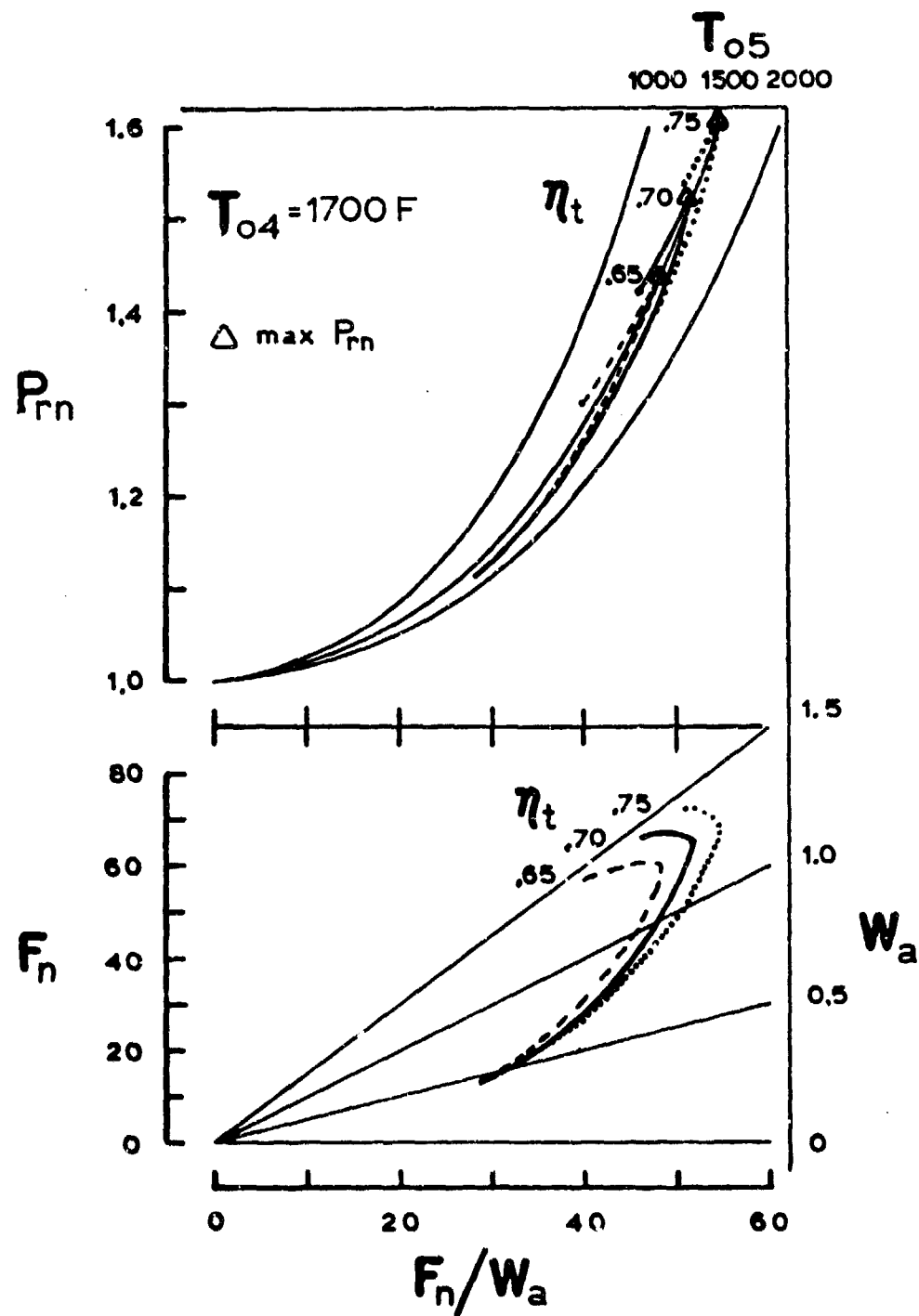


Fig 13: Effect of η_t on Thrust and P_{rn}

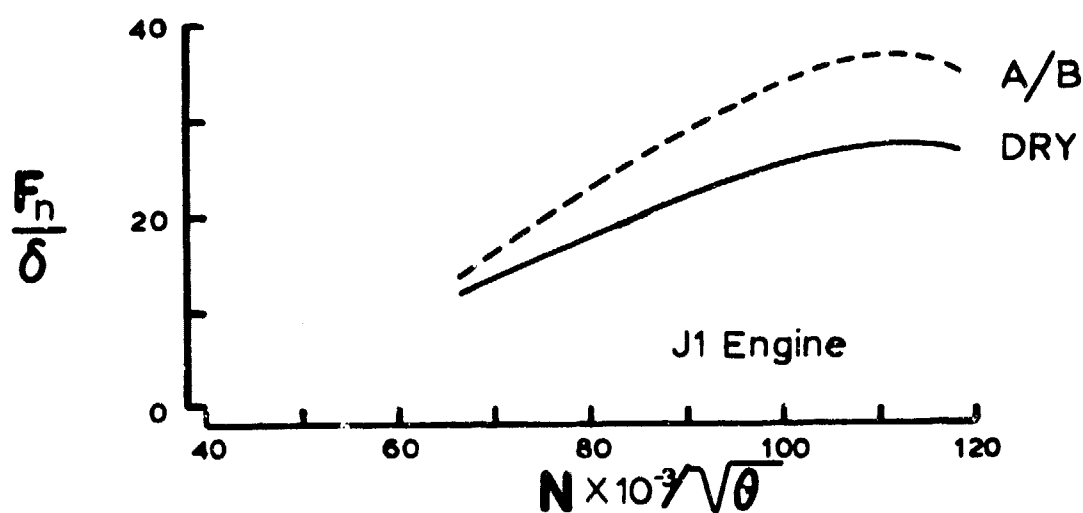
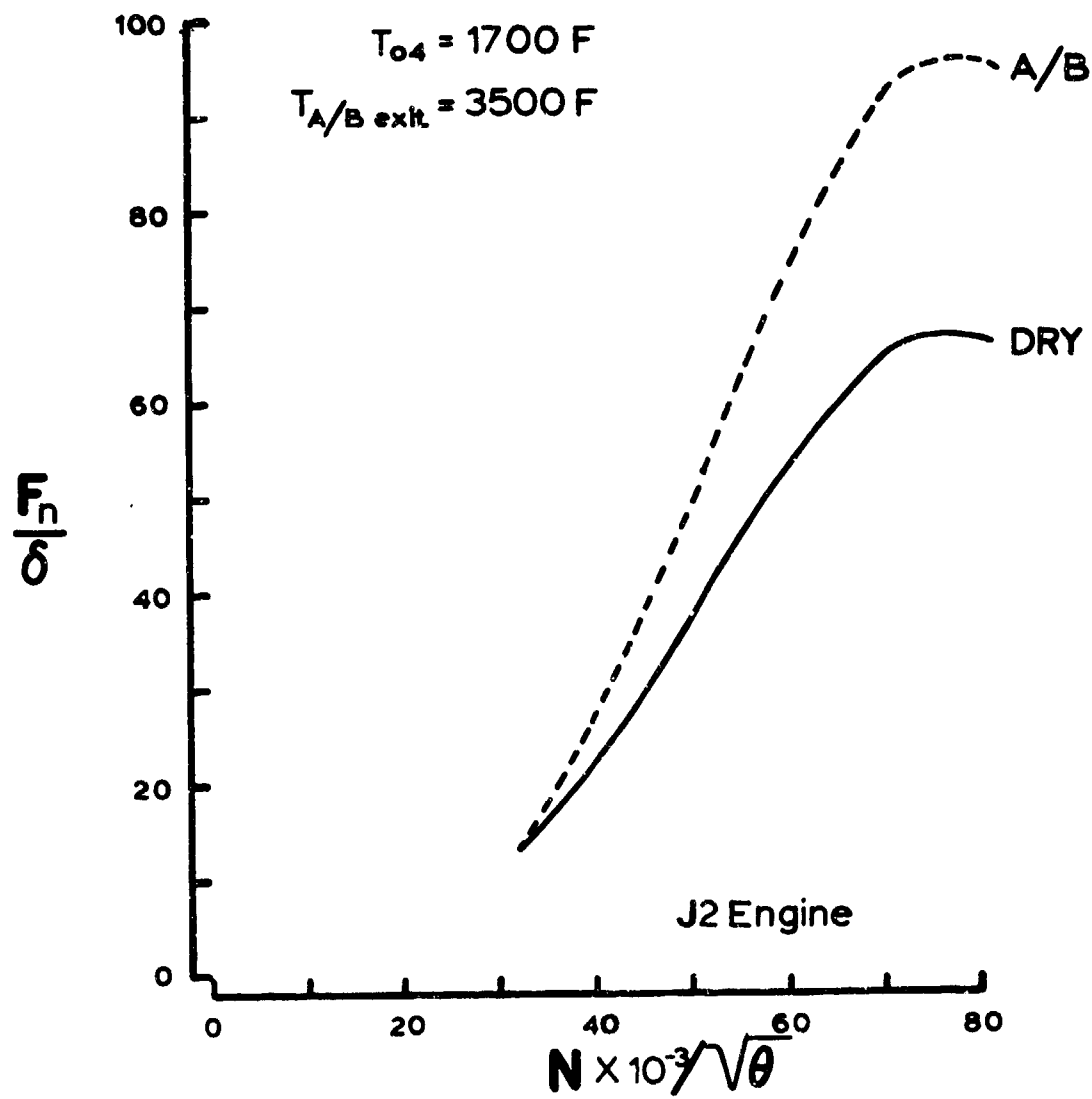


Fig 14: Engine Performance: Augmented and Dry Thrust in in Corrected Variables

to analyze performance based on the efficiency parameters so this approach was not attempted. The thrust equation was used to calculate thrust from temperature and pressure measurements in the nozzle. Also, a device to measure thrust directly was used with the constant area nozzle.

IV. Experimental Results

The results obtained by testing the turbocharger engines were close to the predicted values. Stable thrust levels up to 57 lb with the J2 Engine were measured and the turbocharger engines were operated through a wide range of stable conditions. As is common practice, test data was corrected to standard conditions and plotted to illustrate the results of the investigation and for comparison to the performance analysis. Results were obtained in the following areas:

1. Engine stability
2. Performance of various Turbine Housings
3. Air-flow delivery rate
4. Effect of small changes in the combustion chamber and exit nozzle
5. Development of the lubrication system.

Engine Stability

Stability was the most important performance parameter. Until stable operation could be maintained, data taken was of little use. Engine thrust was low and the bearings did not survive continuous surge loading (see Lubrication System, page 35). Considerable periodic back flow of air from the front of the J2 Engine was observed during unstable operation. A tuft of cloth, tied close to the bellmouth in front of the compressor reversed direction with each pressure pulse (engine surge). When the MA-1A fuel nozzle was in use, engine surge, although less likely to occur, was more violent and flame from the combustion chamber could be seen in the inlet bellmouth. Stable operation was maintained over a wide range

of P_{rc} but maximum thrust was limited by the onset of engine surge.

Turbine Housings

Stable operation was maintained with #4 and #5 turbine housings. Housing #4 provides a 13% increase in maximum thrust over housing #5 for the J2 Engine (see Fig 15). Stable performance data could not be obtained with #1, #2 or #3 housings. The results shown on Fig 15 are data points taken at the peak P_{rn} that a housing could produce. Engine operation was unstable for the data point with the #3 housing.

Air-Flow Delivery Rate

The test results indicate a higher mass flow rate of air (W_a) was delivered by the compressor than determined in the performance analysis. Fig 16 shows this trend for the J1 Engine and Fig 17 for the J2 Engine. As a result, engine thrust was higher than predicted at the compressor pressure ratios where the engine was operated (see Fig 18).

The RPM scales shown on Figs 16, 17, 18 and the other curves in this section is approximate, taken from the performance analysis in Appendix B, although the shape of the curves on the compressor map indicate the scale is within 5%.

Combustion Chamber And Exit Nozzle

Combustion Chamber. The combustion chamber performance was a significant factor in overall engine performance. This is illustrated by the results of a change for the J2-4 Engine from a furnace fuel nozzle to a MA-1A fuel nozzle as shown Fig 19. This change resulted in a 60% increase in thrust and improved the stability of the engine. As another illustration,

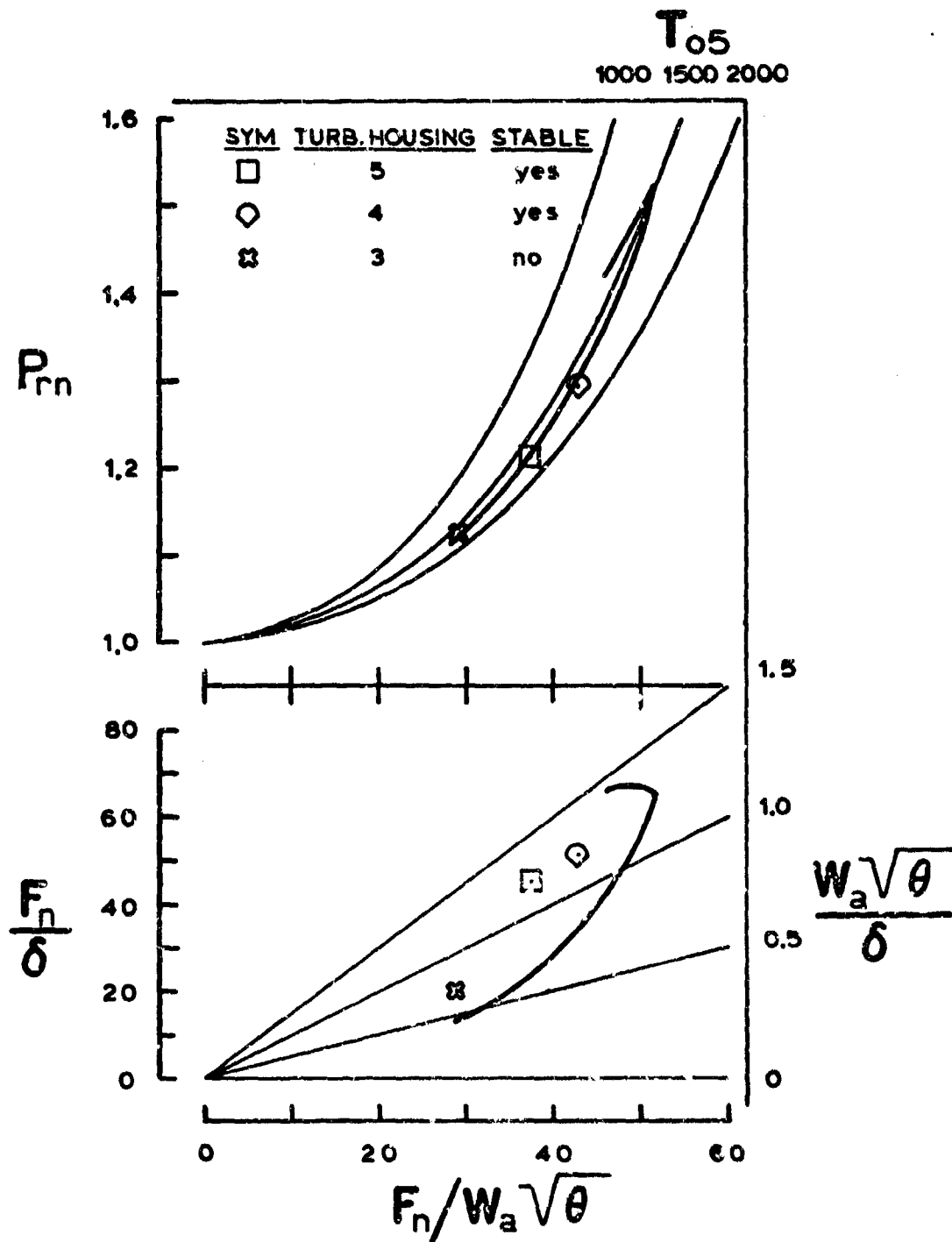
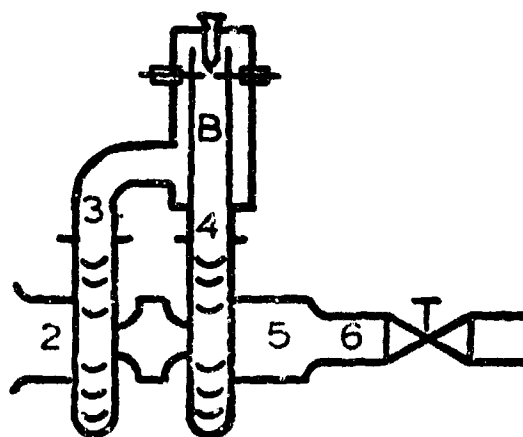


Fig 15: Turbine Housing Performance



J1 ENGINE

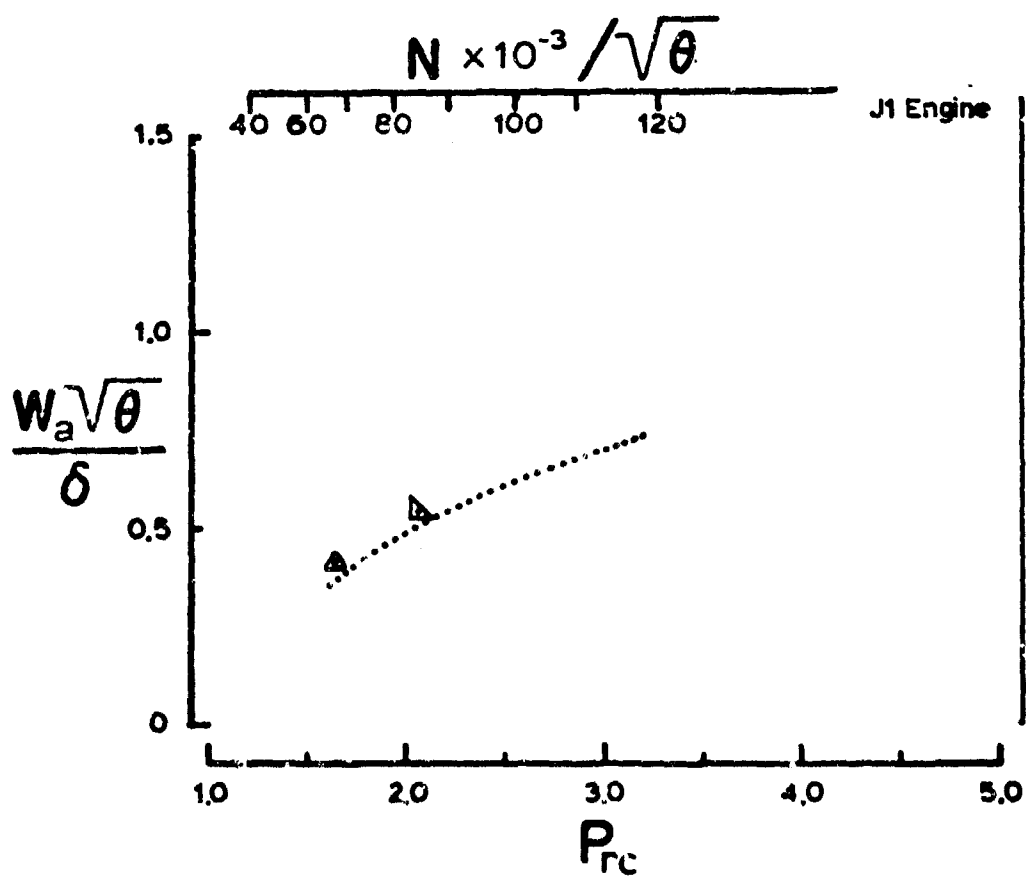
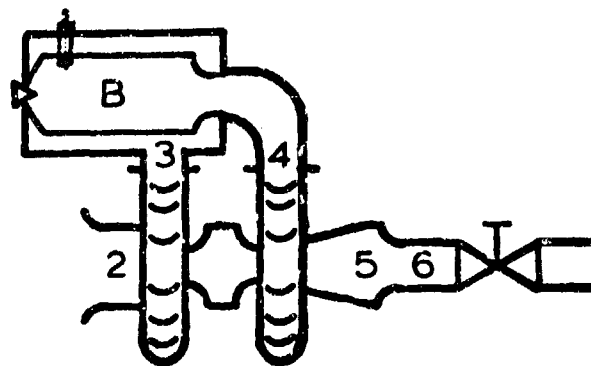


Fig 16: Air-flow for the J1 Engine



J2 ENGINE

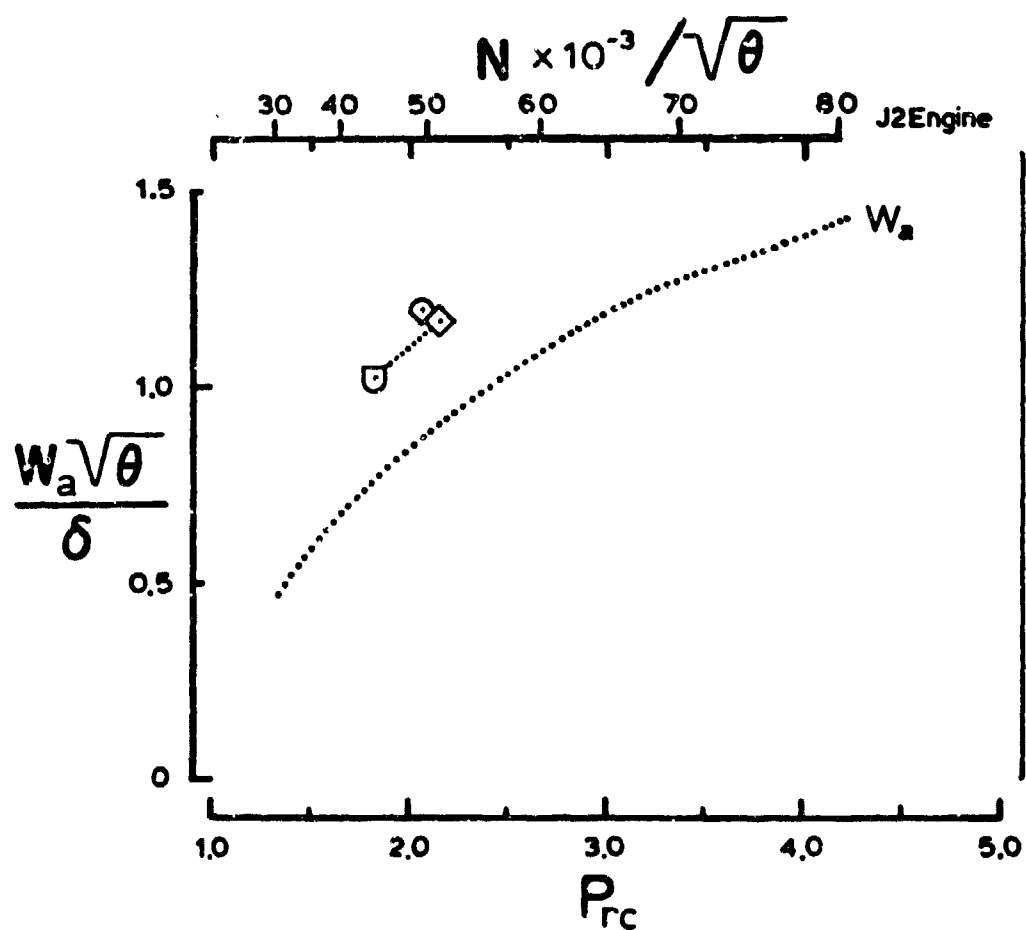
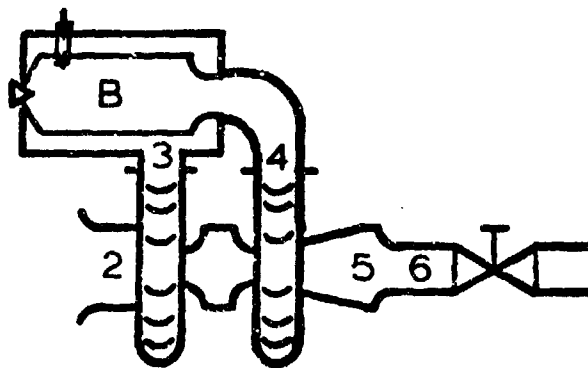


Fig 17: Air-flow for the J2 Engine



J2 ENGINE

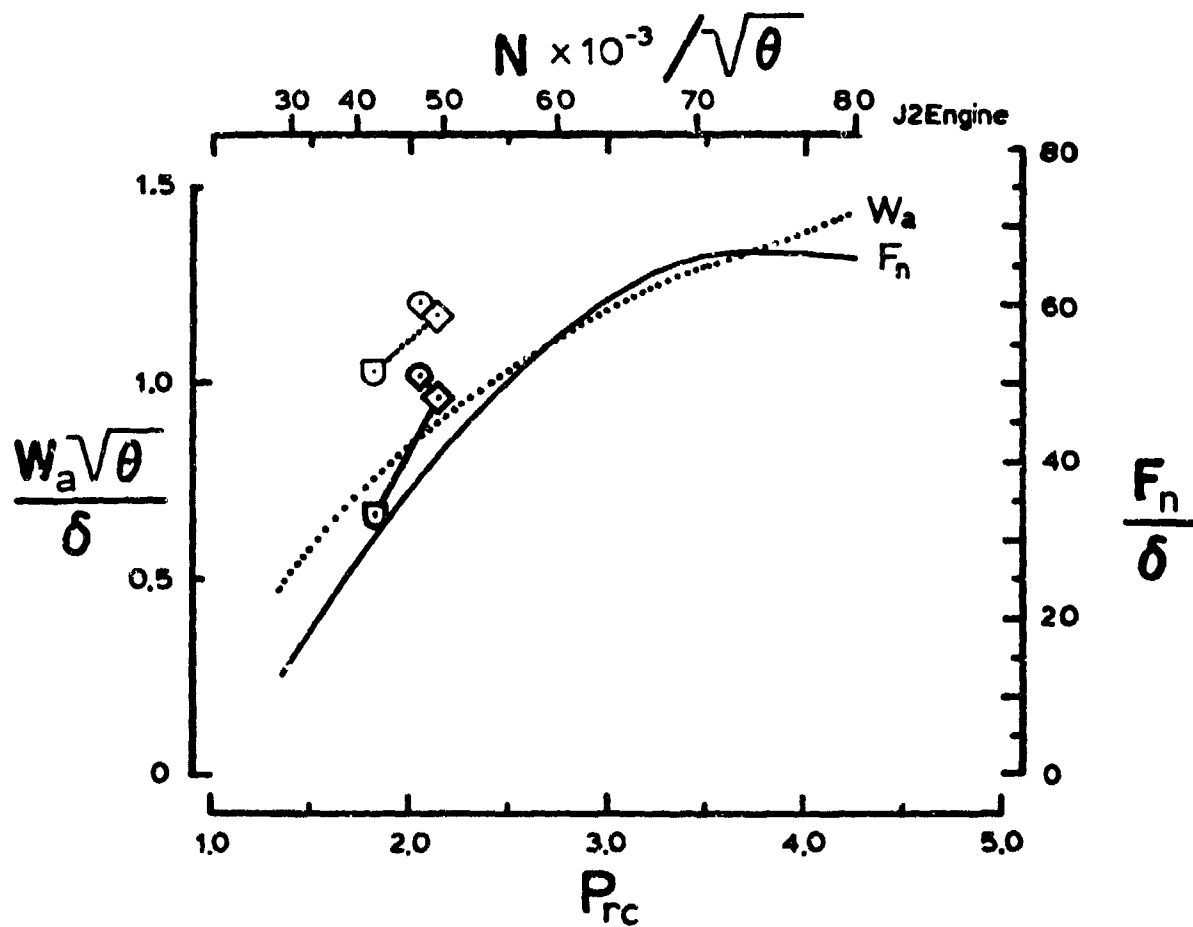


Fig 18: Increased Air-flow Causes Higher Thrust

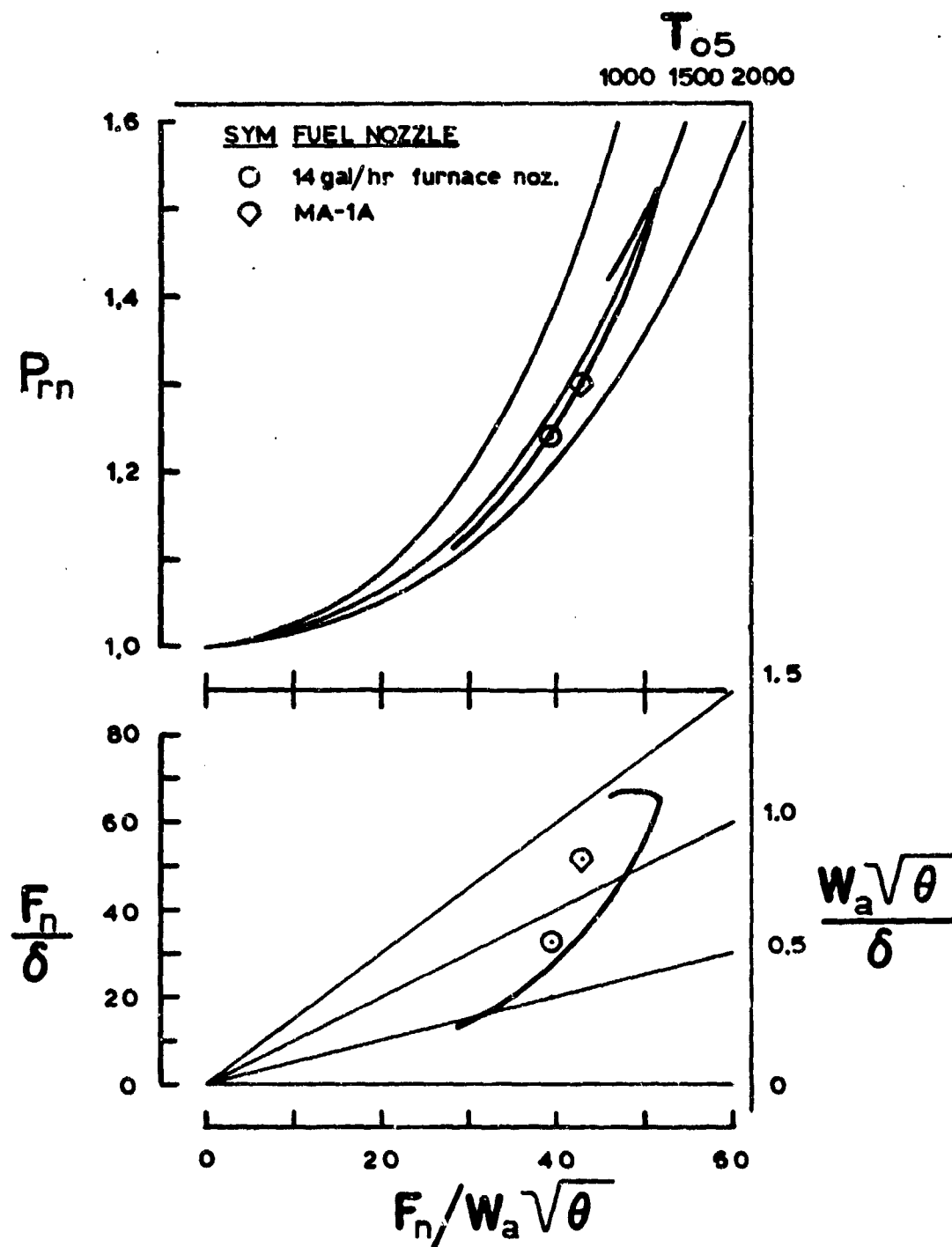


Fig 19: Combustion Chamber Performance

an eight gal/hr furnace nozzle was used initially for the J1 Engine. There was inadequate control of the fuel flow rate at the lower W_a values during engine start-up causing very high gas temperatures at the combustion chamber exit. A four gal/hr furnace nozzle was then installed that gave an adequate peak P_{rc} , satisfactory control and a combustion chamber exit temperature below 2000 F as the engine was started.

The fuel nozzle calibration curves are in Appendix C. For a more detailed discussion of experimental fuel nozzle and combustion chamber results, see Greene (Ref 2).

Exit Nozzle. For the last test runs of the J2-4 Engine, the fixed area engine exhaust nozzle described in Section II was substituted for the gate valve nozzle assembly. This nozzle had no provisions for measuring nozzle inlet pressure or temperature, instead a cantilever deflection beam with strain gage instrumentation was used to measure thrust directly. The three data points taken on one of these runs are plotted on Fig 20 in large symbols. Three of the data points taken earlier with the other nozzle assembly are also plotted on this figure in small symbols for comparison. The engine produced more thrust as W_f was increased. With marginal stability, 60 lb of thrust was measured, stable operation was achieved with 57 lb thrust, and complete data taken with 54, 43 and 29 lb of thrust (see Fig 22, Appendix A).

The gases leaving the turbine to exit the exhaust nozzle appeared to have a large swirling velocity component. They left the nozzle in a nonuniform stream that was not

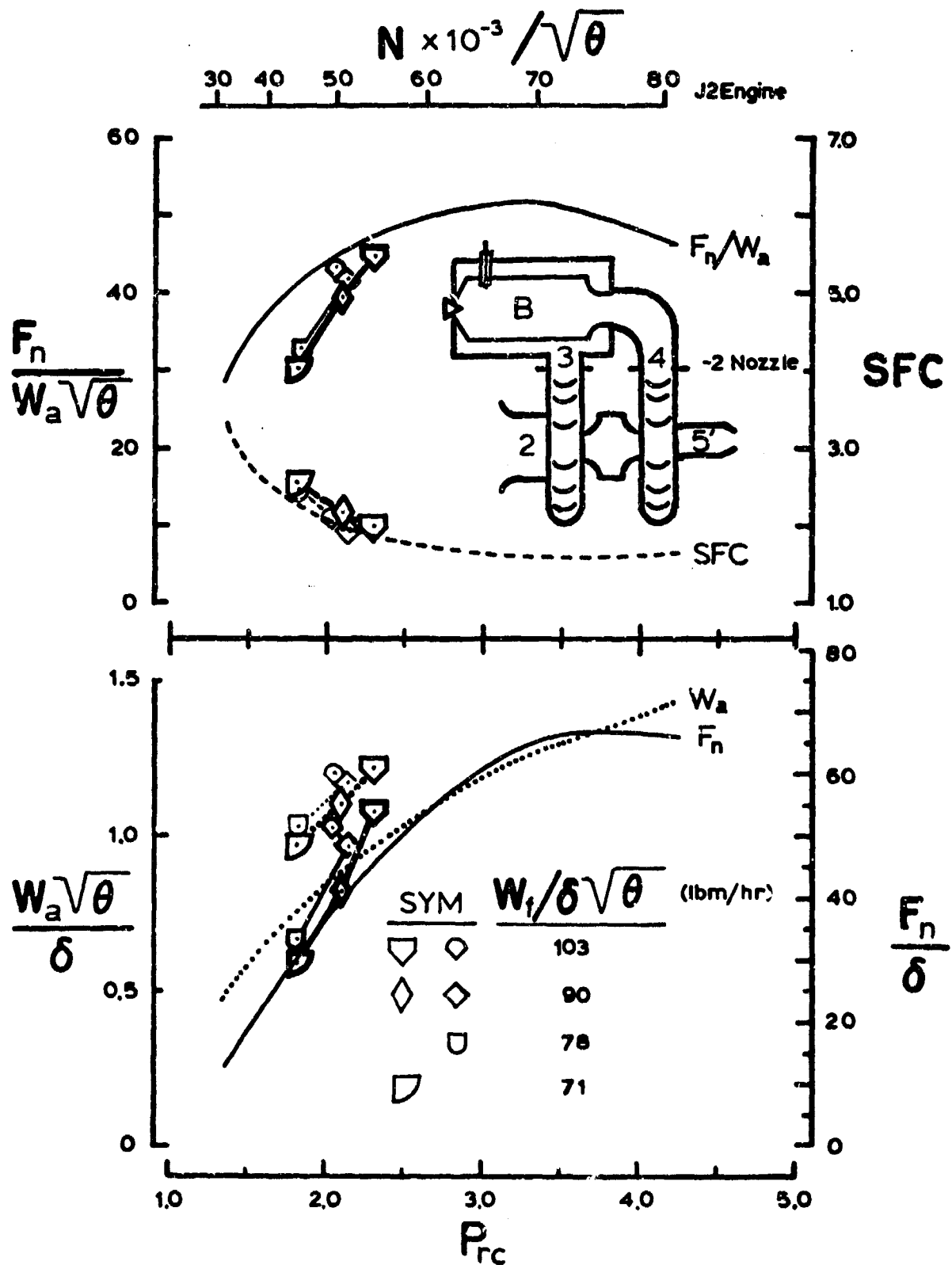


Fig 20: Exit Nozzle Performance from Direct Thrust Measurement

parallel to the engine centerline at low W_a , but became more aligned as W_a increased.

Engine exit nozzle size is critical for proper engine operation. All the engine components must be matched to obtain maximum performance; however, in these engines the nozzle regulates the total air-flow because the turbine nozzle is not choked. The effect of improper nozzle sizing was shown by removing the valve from the nozzle assembly on the J2-4-1 Engine. This formed a nozzle 25% larger than that used during the successful J2-4-2 Engine test described above, and the engine could not be stabilized with the larger nozzle in place.

Lubrication System

The lubrication system was improved to the point where performance data could be taken with the engines over a long period of time with no bearing deterioration. Approximately 30 minutes of running time were accumulated on the J2A Engine, and 38 minutes on the J2B Engine. The J2B Engine was showing no signs of deterioration at this point.

Approximately five minutes of running time was obtained with J1 Engine before bearing failure. Little numerical data was obtained with this engine; however, the test resulted in check out or significant improvement of starter design, the ignition system, instrumentation system, oil system, and the engine support structure. When the bearings of the J1 Engine failed, one of the contributing factors seemed to be hot gases forced into the bearings by high turbine pressure.

The bearings in the J2A Engine had failed after a series of tests using turbine housing #3. Pressure surges increased the back-flow of hot gases during unstable J2 Engine operation. A surge of gas and oil in the oil return lines when using the engine starter was an indication of impending bearing failure. The oil return system was pressurized by bleed air from the compressor during the J2B Engine test to reduce the flow of hot gases into the bearings; however, this forced oil into the turbine and exhaust nozzle assembly causing afterburning and unstable engine operation.

Inspection of the J2A Engine bearings indicated failure due to high loading. Also the forward bearing was more heavily damaged than the aft bearing (close to the turbine). Twenty minutes running time was obtained with the J2B Engine after pressurization was discontinued with no apparent bearing damage. These results indicated bearing damage had been due to the surge loading, not hot gas back-flow, and pressurization was not necessary for satisfactory operation.

Discussion Of Performance Trends

Turbine blades on the J1, J2A and J2B engines showed no sign of deterioration from the high temperature gases. For most of the runs, peak temperatures entering the turbine housing were approximately 2000F or as limited by engine stability (see Appendix A). The largest continuous run was for seven minutes with an indicated T_{04} of 1960F. The temperature was measured with a shielded thermocouple positioned at the peak temperature point of the combustion chamber exit.

See Greene (Ref 2) for a more complete discussion of turbine entry temperature and Appendix C for temperature measurement accuracy.

Performance trends can be analyzed by referring to Fig 21, which shows the performance data on an abbreviated compressor map.

1. Smaller turbine housings move the operating line to the left (Compare J2-5-1 to J2-4-1 Engine) increasing the P_{rc} and thrust but decreasing surge margin as the operating line moves close to the surge limit.

2. Because of small instabilities caused by interaction of the engine components, the surge limit has been moved to the right and is close to where the operating line was assumed. The surge limit is between the J2-3-1 (unstable) and the J2-4-1^a data points.

3. The figure indicates W_a is approximately 25% higher than is indicated by the assumed operating line. The actual operating line is to the right of the assumed operating line at these low P_{rc} levels. The measured thrust (see Fig 20) is 17% higher than the predicted values.

4. Large performance gains were possible by improvement in the compressor-turbine match (J2-5-1 to J2-4-1) or by improving the combustion chamber (J2-4-1^a to J2-4-1) and the exit nozzle (J2-4-1 to J2-4-2).

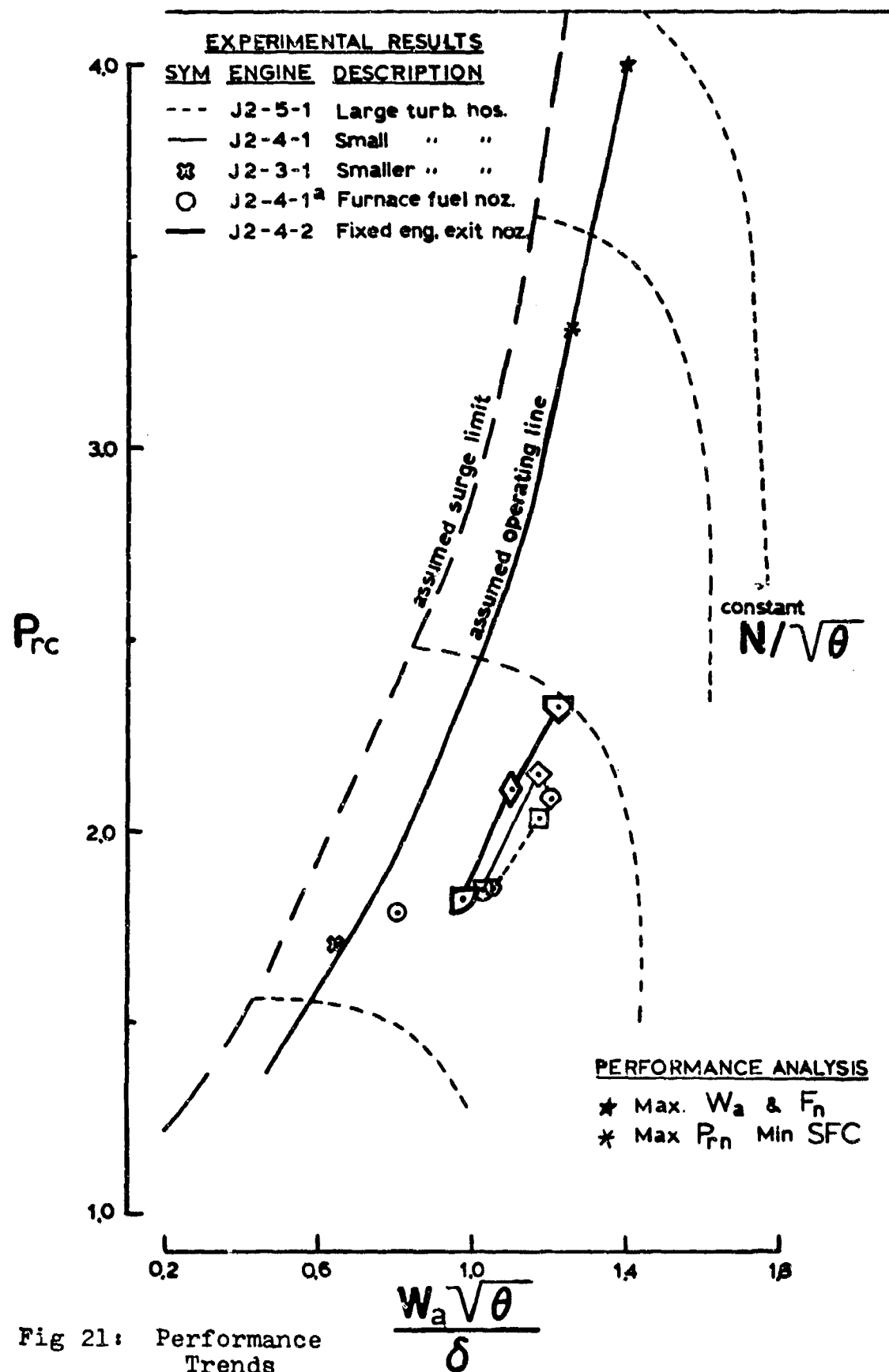


Fig 21: Performance Trends

V. Conclusions And Recommendations

Conclusions

The objective of developing a turbojet engine from a turbocharger was achieved as both engines designed and tested were self-sustaining units on the test stand. Sixty lb of thrust was generated by one of these engines as its thrust was increased by various improvements to within 10% of the maximum predicted performance. The large effect of temperature on performance (see Appendix B) indicates significant thrust increases would be possible over the maximum predicted by increasing the turbine inlet temperature.

Recommendations

Several recommendations requiring little change in the test equipment can be made. Other recommendations requiring major redesign of components will require more time and effort. For the engines tested, improvement or redesign of the combustion chambers, engine nozzles and turbine housings, provide the greatest potential for better performance.

Short Term. Improvements can be made in the combustion chamber stability and mass flow capability. A series of nozzle tests could improve performance by better sizing and the use of flow straighteners. With improvement in engine stability, one of the other turbine housings may show better performance than the best one from this investigation. Also, the turbine may be able to tolerate higher temperatures than were used in this study. The J1 Engine should be tested in

more detail to confirm the results obtained in this investigation.

Long Term. Transducers should be incorporated in the instrumentation system in order to simultaneously record all data to improve the accuracy of data acquisition. Detailed temperature profile and mass flow studies to determine average temperature from the peak temperature measured in this investigation would be helpful. For more detailed studies, the error in the thrust measurement system should be reduced. Also, a device to measure RPM would be a necessity.

A weight reduction study for this engine will be necessary to produce a machine suitable for installation in an aircraft. The weight of the turbine housing (see Table II) amounts to almost half of the total machine weight. Also significant increases in performance by increasing the P_{rc} should be possible with a better turbine-compressor match. A redesign of the turbine housing could reduce the weight to around seven lb and give a better match. An analytic study of compressor-turbine matching for these engines would be helpful in the design. A series of tests using fuel in place of oil for lubrication and cooling of the bearings should be considered. Also, isolation of the engine components by using a separate air source to supply air for the combustion chamber and turbine while measuring compressor output should be helpful.

The performance analysis indicated considerable performance gains were possible with thrust augmentation. An

afterburner and other forms of thrust augmentation such as water injection should be tested on the engines.

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APPENDIX A

TEST DATA

Table V shows the most important data points obtained from the experimental investigation. This data has been corrected to standard conditions. An explanation of the symbols used is as follows:

- P_t - data point number
- SYM - symbol used to plot this point in the other section of this document
- ENG - engine number
- f_{noz} - fuel nozzle used for the data point
- W_a - lb/sec air-flow
- P_{rc} - compressor pressure ratio
- T_o - total temperature F (subscript denotes station number, see Fig 8)
- P_{rn} - pressure across the nozzle (P_{o5}/P_{∞})
- F_n - thrust
- SFC - specific fuel consumption (lbm fuel per hr/lbf thrust)

The letters in the table refer to the following notes:

- ^a - gal/hr furnace nozzle
- ^b - 14^a and MA-1A were tried, performance about the same and all were unstable
- ^c - only thrust data taken
- ^d - approximate
- ^e - data not available with this exit nozzle, F_n measured directly
- ^g - nozzle instrumentation not installed

TABLE V. TEST DATA

NOTE: The F_n vs. time plot for data points 1-5 is shown on Fig 22

Pt	SYM	ENG	Fnoz	Wa	Prc	To3	To4	Pin	To5	F_n/W_a	SFC	F_n
1	...	J2-4-2	MA-1A	60c
2	...	"	"	...	(2.38)d	57c
3	▽	J2-4-2	MA-1A	1.22	2.31	245	2090	44.6	1.98	54
4	◇	"	"	1.10	2.10	210	1870	39.4	2.18	43
5	▷	"	"	0.97	1.81	210	2010	30.1	2.52	29
6	○	J2-4-1	MA-1A	1.20	2.07	180	1910	1.30	1620	43.0	2.09	51
7	◇	"	"	1.17	2.14	175	1960	1.26	1730	41.4	1.95	48
8	◻	"	"	1.02	1.83	220	2060	1.16	1600	32.4	2.45	33
9	◻	J2-5-1	MA-1A	1.17	2.02	210	1880	1.22	1650	37.9	2.27	45
10	○	"	"	1.05	1.84	205	1700	1.21
11	○	J2-4-1	14 ^a	0.80	1.77	...	2070	1.24	1740	39.6	2.43	32
12	⊗	J2-3-1	8ab	0.64	1.69	164	2070	1.13	1550	29.1	3.31	19
13	△	J1-2-1	4a	0.54	2.06	300	1700
14	△	J1-2-1	"	0.41	1.63	220	1500

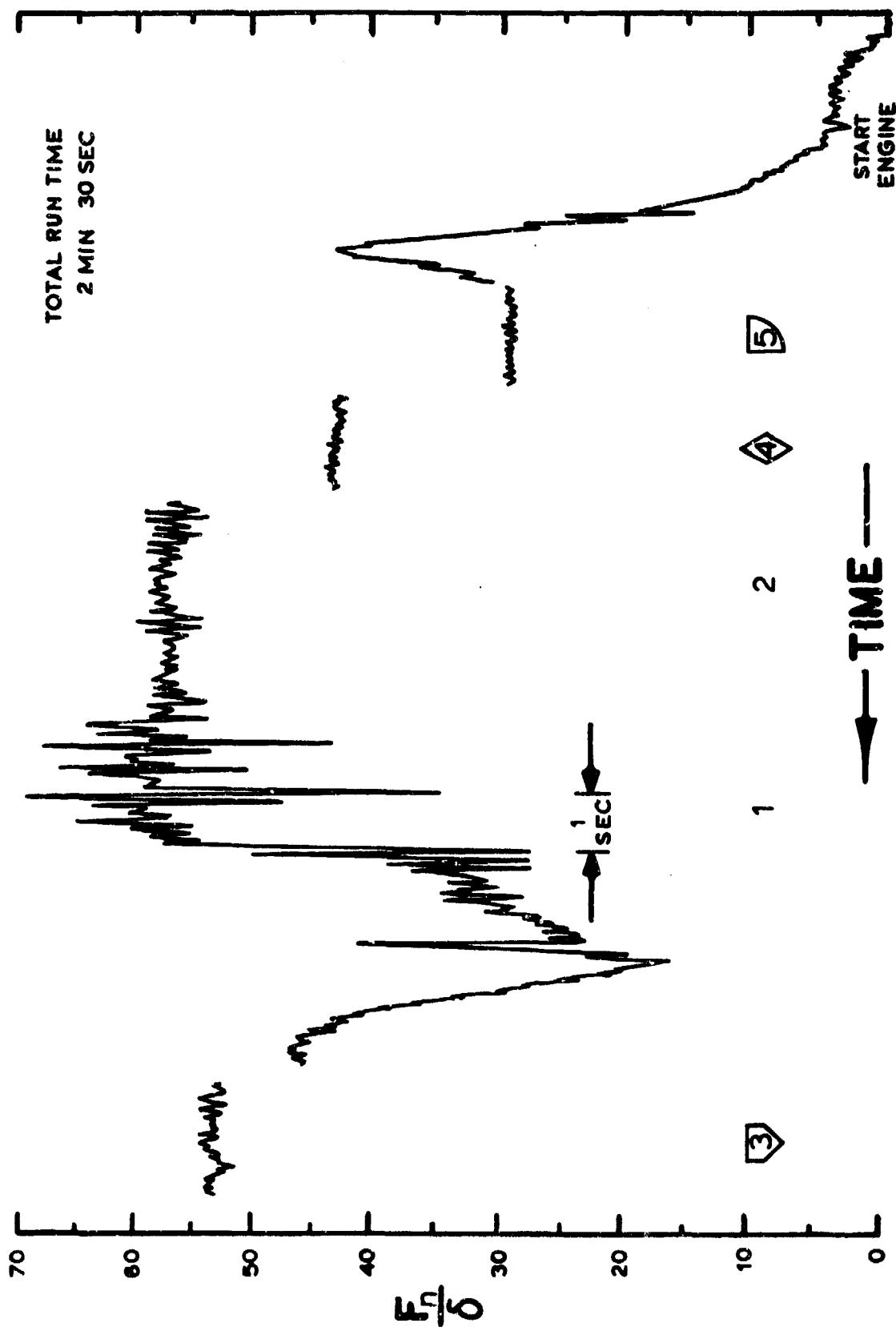


Fig 22: Direct Thrust Measurement Data Sample

APPENDIX B

Performance Analysis

Technique Used

Analytic predictions of engine performance were made using a computer program developed by the Air Force Aero Propulsion Laboratory which uses simple thermodynamic Brayton cycle analysis.

The "Design Point Turbine Engine Performance Program" (Ref 5) called Carpet can predict the performance of an engine from the laws of thermodynamics, given the maximum temperature, compressor pressure ratio, and the appropriate efficiencies of the engine components. Table VI lists the components considered in the simple turbojet analysis.

Table VI.

Turbocharger-Engine Components And Performance Parameters

Component	Parameters
Compressor	P_{rc}, η_c
Combustion chamber	$T_{o4}, \Delta P_{ob}, \eta_b$
Turbine	η_t
Nozzle	η_n^a

^aCarpet uses a velocity coefficient

Three things were accomplished in the analysis:

1. Relative importance of the performance parameters was determined
2. Turbocharger engine performance was predicted for the test conditions

3. Augmented thrust performance was predicted for each engine.

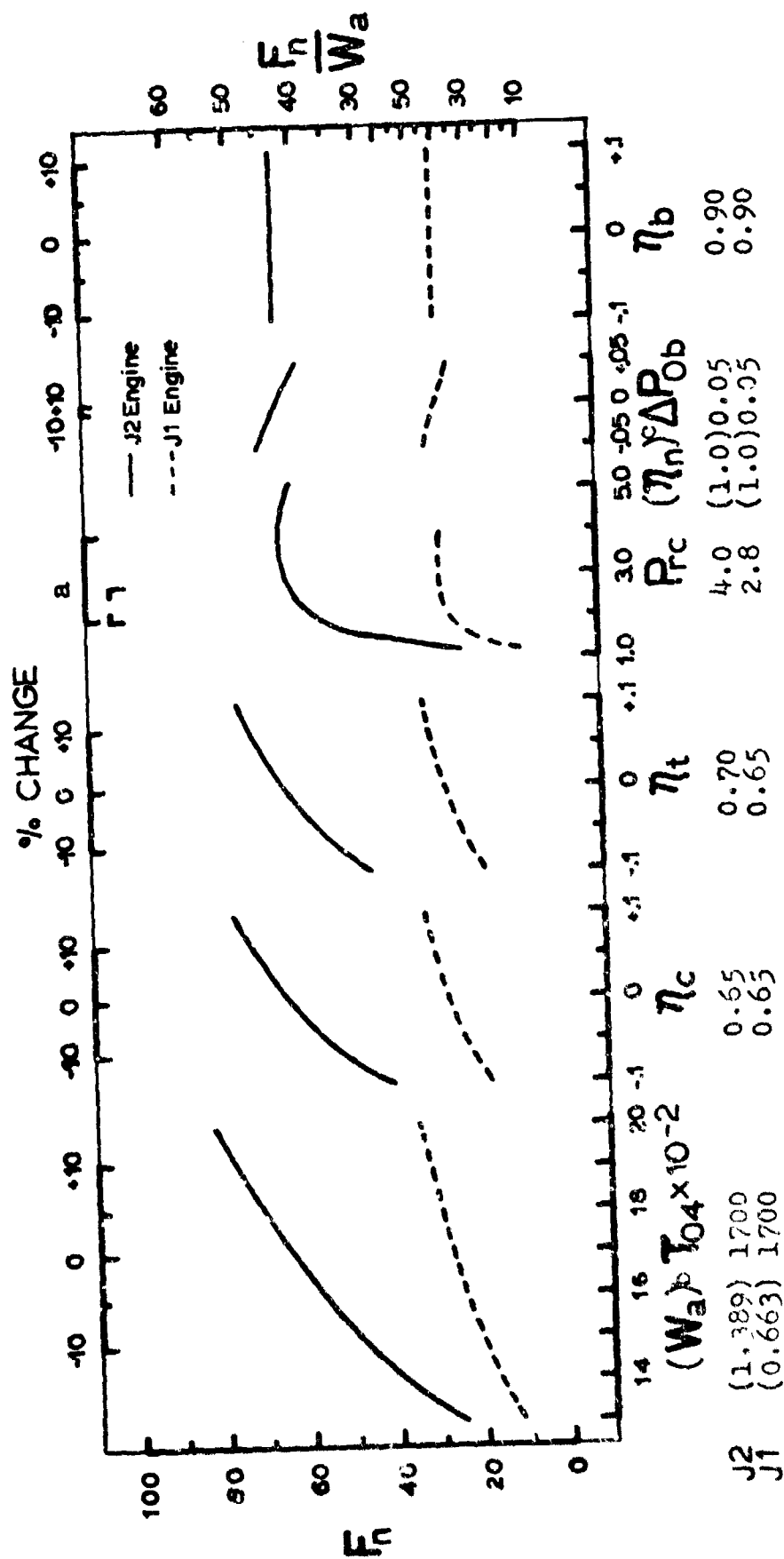
Relative Importance of Parameters

The importance of the performance parameters depends on the mission of the vehicle and on the value of the parameters themselves. Parameters are plotted in order of importance in Figs 23 and 24. The order of importance for most of the variables was determined by comparing the the performance degradation caused by changing the parameter 10% of its base-line value. This base-line value is shown below the plot. The percent change of this base-line value is shown below by the upper scales. Change from this base-line value is shown on the lower scale. For Compressor ratio, a range of acceptable performance is shown on the upper scale; pressure ratio itself is shown on the lower scale. The effect of engine size (W_a) on performance is also shown on Figs 23 and 24. This parameter is determined by comparing the two different engines. Each plot shows the effect of changing a single parameter when the others are held constant.

For these engines, thrust (F_n), shown by Fig 23, is the most important consideration. Specific fuel consumption (SFC) shown by Fig 24 is of secondary importance. Performance parameters in order of importance as shown by Fig 22 are W_a , T_{o4} ,

η_c , η_t , P_{rc} , η_n , ΔP_{ob} , η_b . The effect of nozzle efficiency (η_n) on thrust can be shown to be

$$F_n = \sqrt{\eta_n} F_{n2}(\text{from Carpet}) \quad (2)$$



a Range of P_{rc} for acceptable performance

b Not plotted as % change, compare engines

c See text

Fig 23: Relative Importance of Performance Parameters (Thrust Orientation)

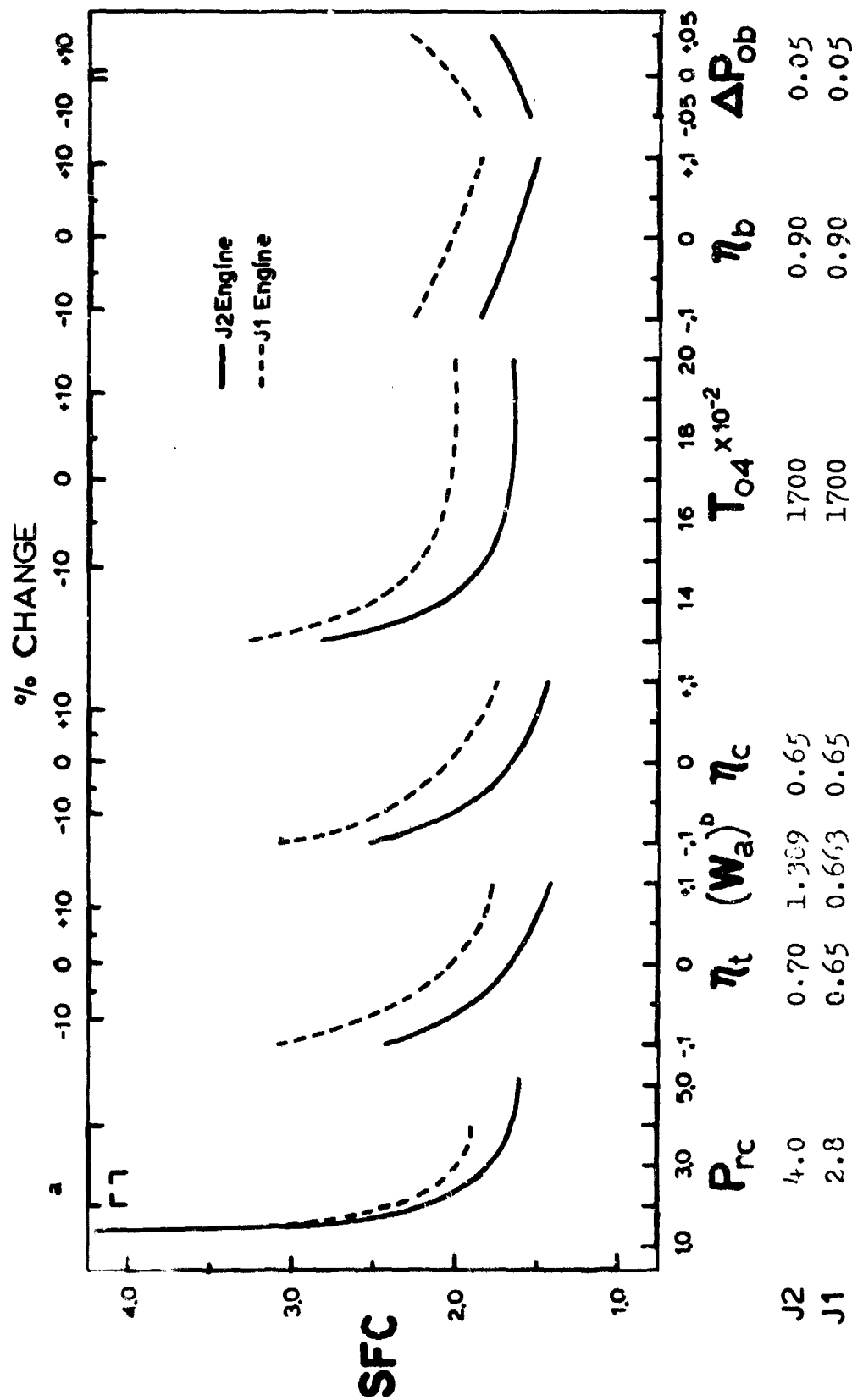


Fig 24. Relative Importance of Performance Parameters (SFC Orientation)

thrust, as calculated by Carpet in this analytic study, assumes expansion through an ideal nozzle to atmospheric pressure. From Eq 1 for $\eta_n = 0.90$ and F_{n2} (from Carpet) = 64.3

$$F_{n1} = \sqrt{0.9 (64.3)} = 61.0$$

Comparing this with the effect of a 10% change in the other variables of Fig 23, η_n is between P_{rc} and ΔP_{ob} in order of importance.

Predicted Performance

The performance that could be expected from the engines operated on the test stand was computed and plotted for the J1 Engine on Fig 25 and as Fig 26 for the J2 Engine. The manufacturer's compressor maps were used in these calculations for W_a , η_c , and P_{rc} . An expected range of performance is shown for the turbine. All other parameters are as shown on Fig 23 and 24 as base line values, so the predicted performance is as determined by these assumed parameters and an assumed operating line (maximum η_c) on the compressor maps (see Fig 10 for typical map).

The maximum thrust point occurs at a higher P_{rc} than maximum specific thrust if η_t is high enough. This is shown in more detail on Fig 27 where the primary variables in the thrust equation from Ref 6 are plotted. In this equation

$$F_n / W_a = (1 + f) \sqrt{2 C_p T_{05} \eta_n \left[1 - \left(\frac{1}{P_{rn}} \right)^{\frac{\gamma-1}{\gamma}} \right]} \quad (1)$$

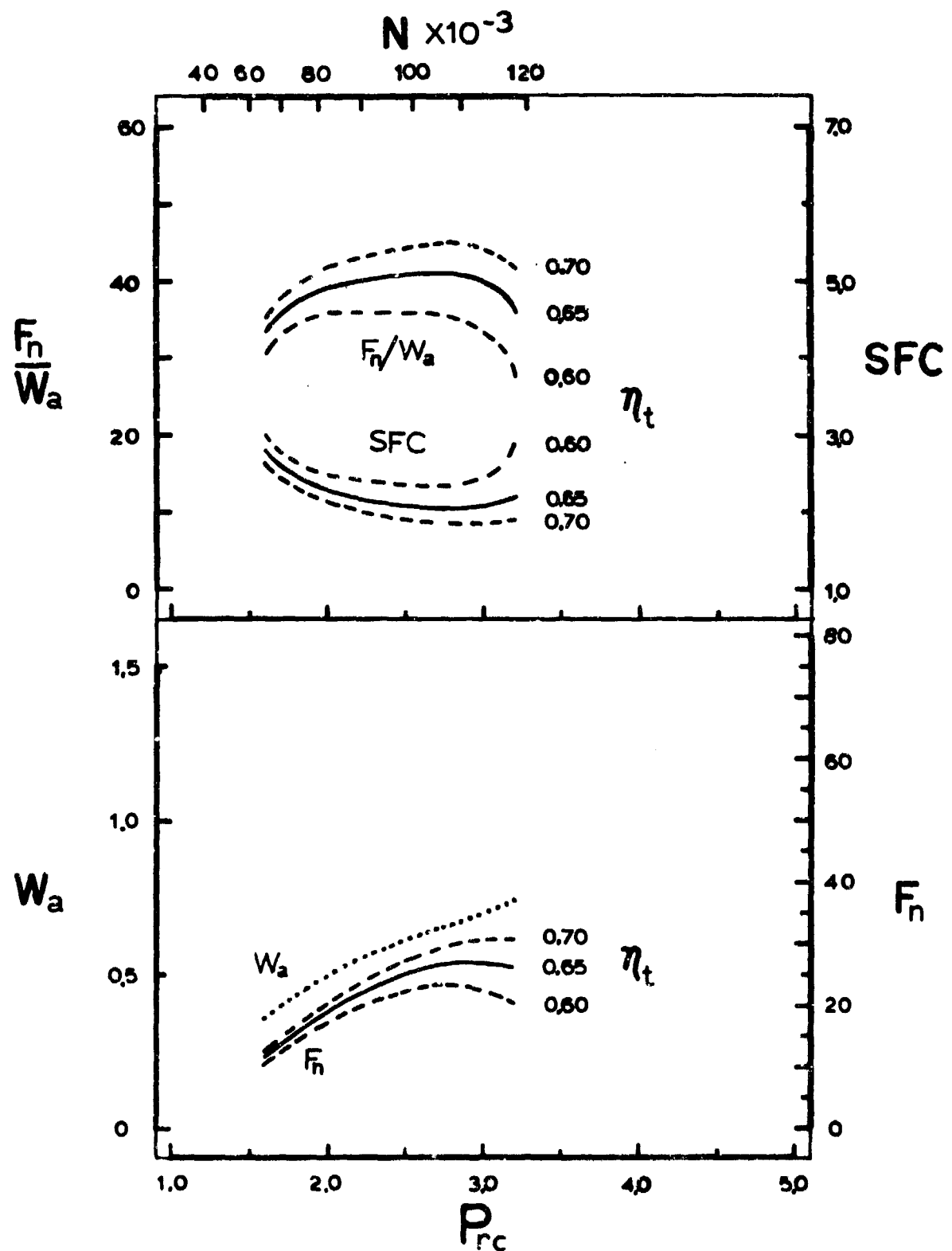


Fig 25: Theoretical J1 Engine Performance,
 $T_{o4} = 1700F$

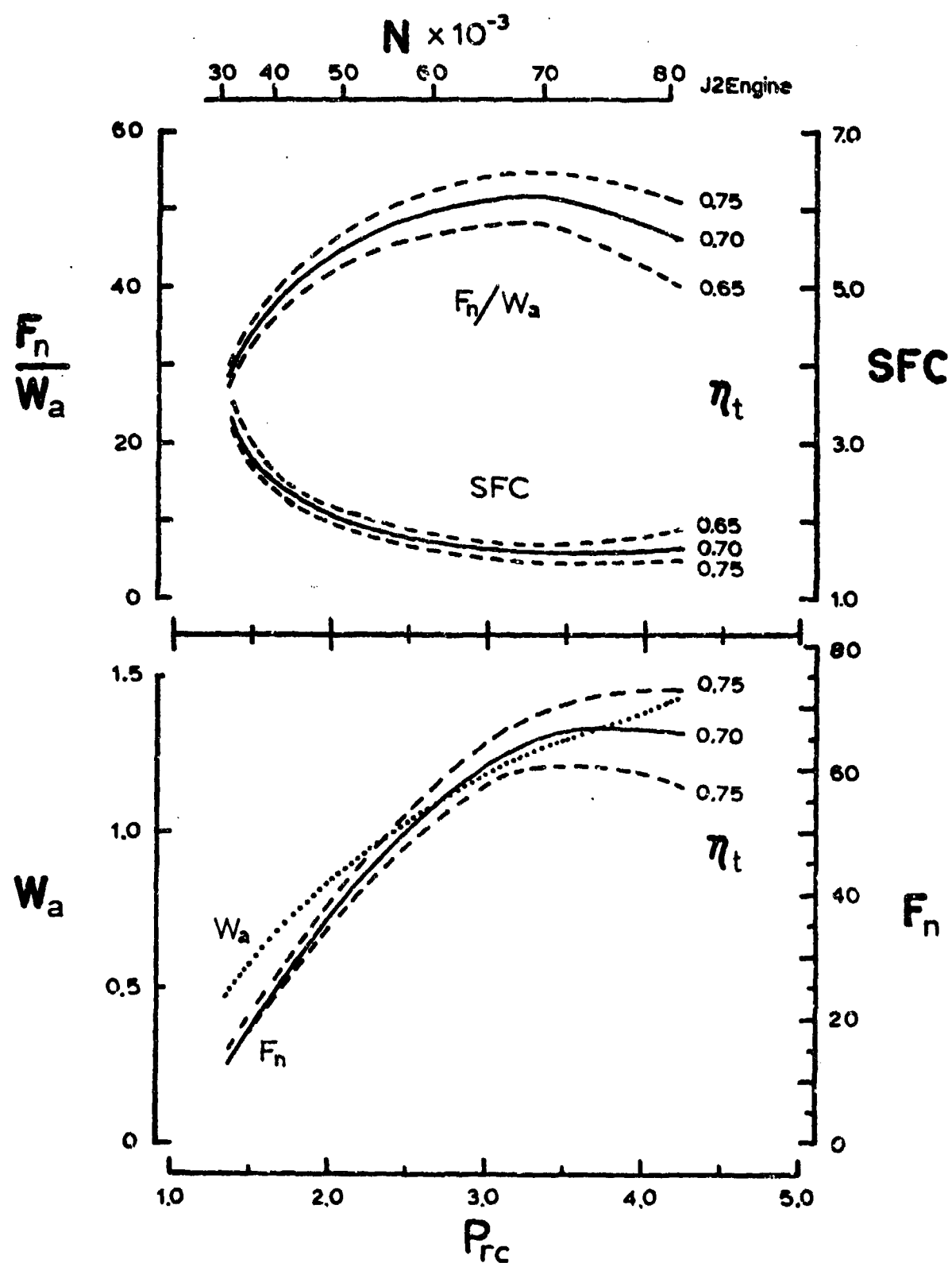


Fig 26: Theoretical J2 Engine Performance, $T_{o4} \approx 1700$ F

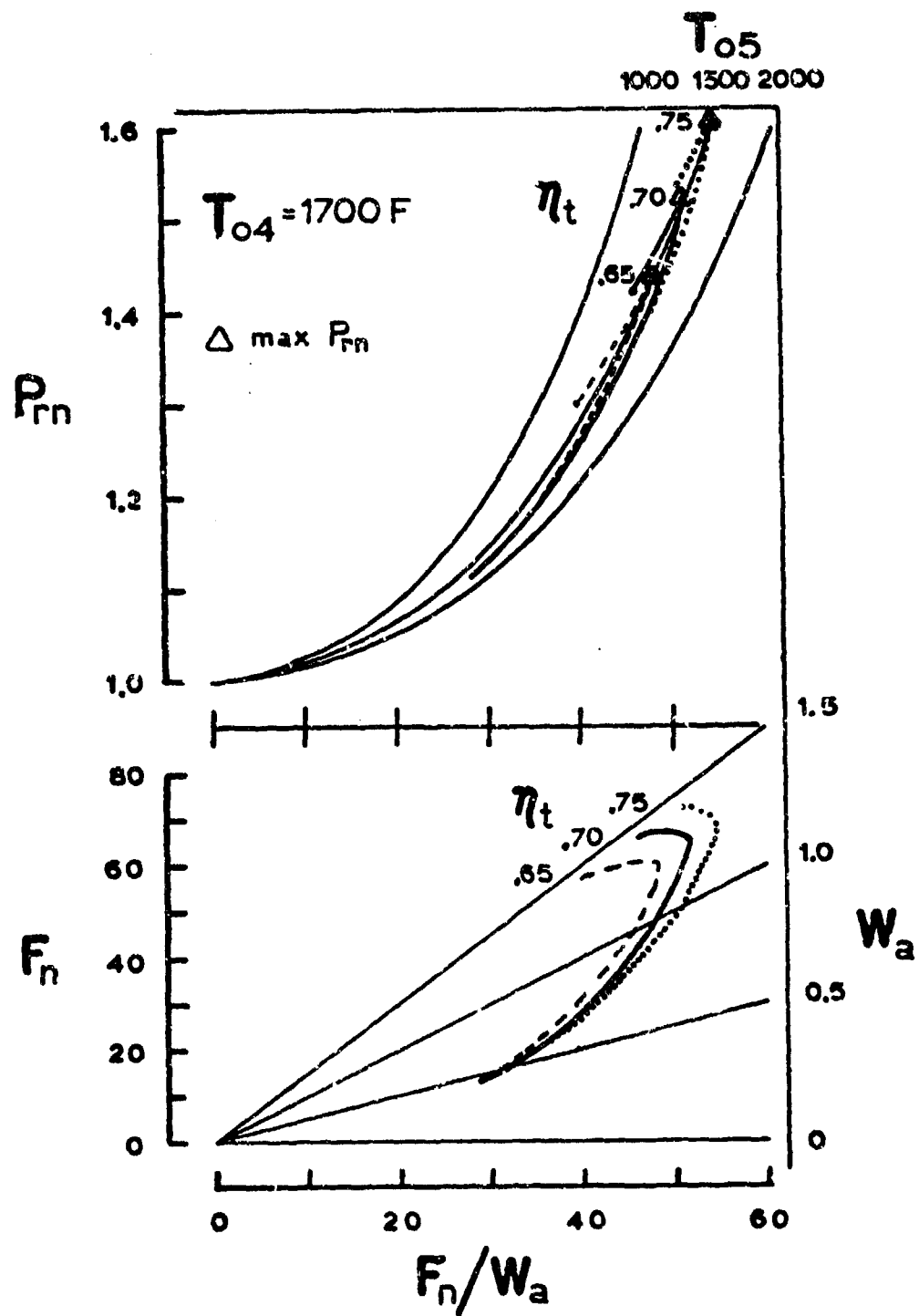


Fig 27: Theoretical J2 Engine Thrust with Major Parameters

two of the variables, $f = .02$ and $\eta_n = 1.0$, were assumed constant. The other two variables C_p and γ depend on T_{05} and are from Thermodynamic Gas Tables (Ref 7). The maximum specific thrust does not depend on W_a and occurs where P_{Tn} is maximum. However, an increase as may be seen for $\eta_n = 0.7$ and 0.75 .

A comparison of predicted performance for the two engines is shown in Fig 28. The curves for the engines are very similar in shape. The J2 Engine allows operation at higher compressor pressure ratios, mass flows, and a higher thrust to be produced. The higher turbine efficiency assumed for this engine (see Fig 23) also increases its performance.

Augmented Thrust Performance

Predicted performance of the engines with afterburners is shown on Fig 29 for an assumed afterburner exit temperature of 3500F. A total-pressure loss in the afterburner of 5% and a combustion efficiency of 90% was also included.

Results

There were a number of important results in the performance analysis. A significant amount of difference occurs in the effect of the performance parameters on engine performance. A reasonably high thrust can be produced with turbo-charger engines and the thrust level can be augmented significantly by afterburning. Also, some of the performance parameters will be less affected by instrumentation error than others.

Three parameters η_n , ΔP_{ob} and η_b have much less effect on thrust than the others in the investigation.

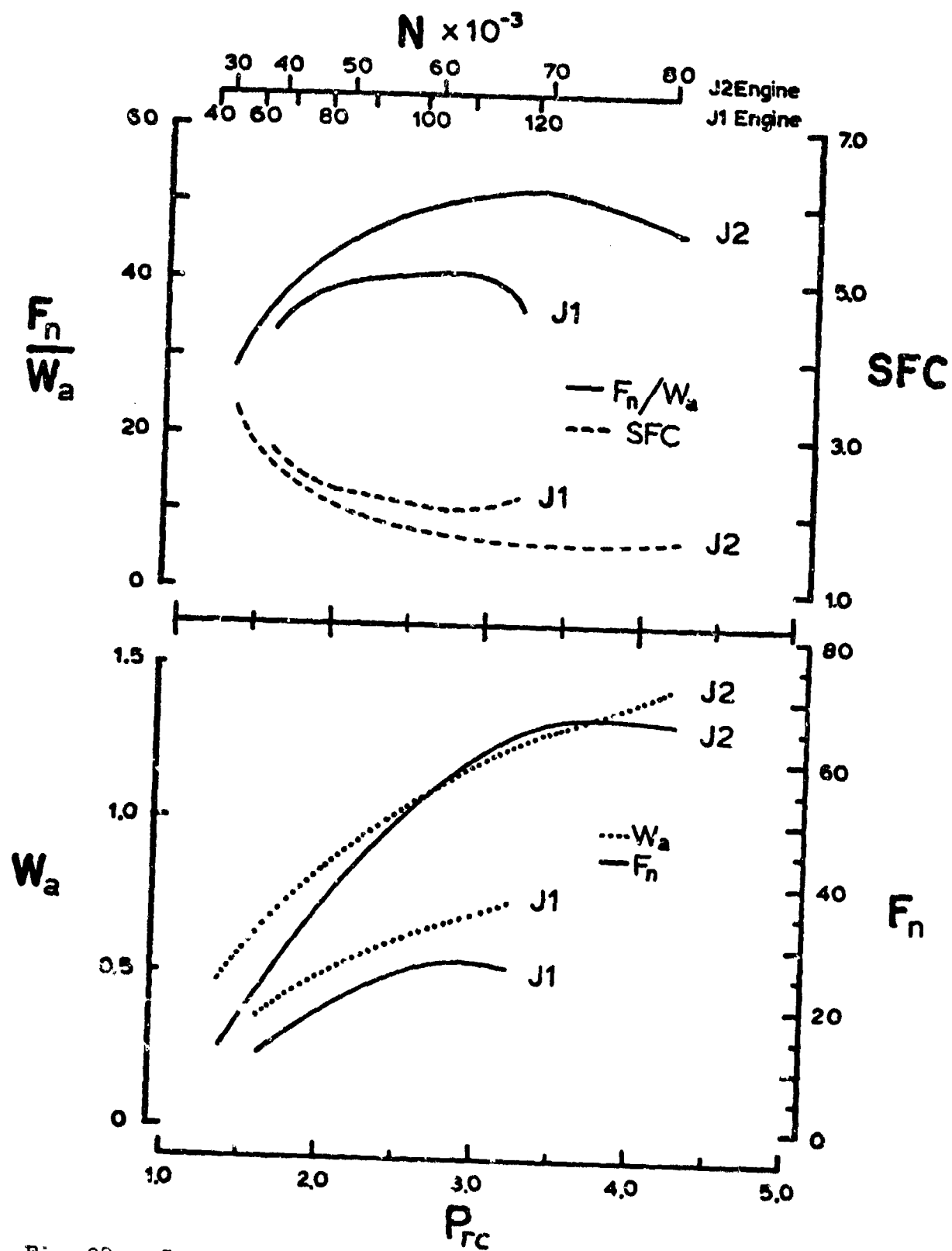


Fig 28: Comparison of Theoretical Engine Performance

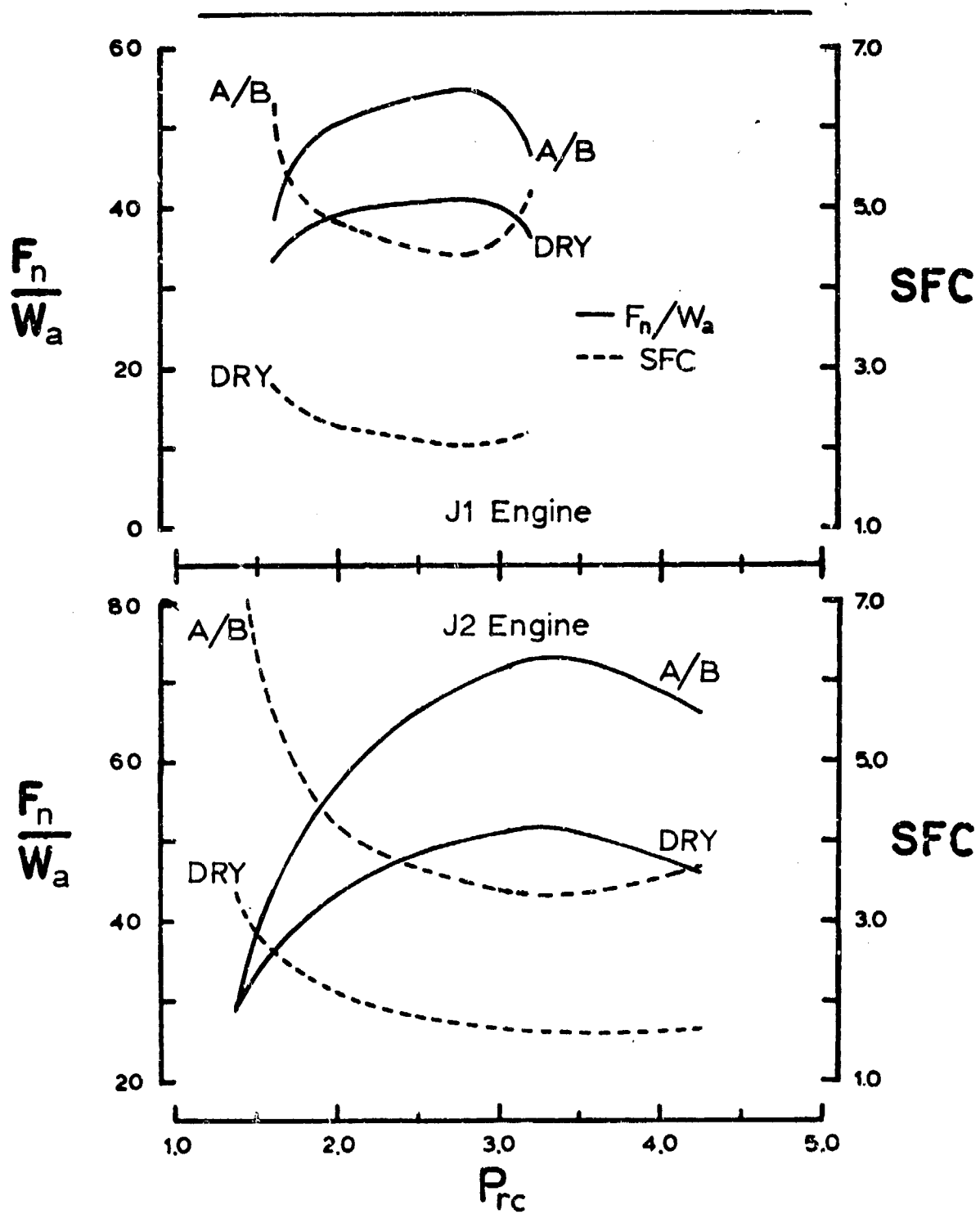


Fig 29: Specific Augmented and Dry Performance

The maximum thrust expected from the dry engines, with the assumptions made in the performance analysis, is 27 lb for the J1 Engine and 67 lb for the J2 Engine. For both engines a significant amount of thrust is produced for $P_{rc} \geq 2.0$ and close to the maximum thrust can be produced over a wide range of P_{rc} (see Figs 25 and 26). There is no advantage in working the compressor to a higher P_{rc} than that for maximum thrust, and for the J2 Engine a slightly lower thrust level (65 lb at 70,000 RPM) was desirable.

Thrust augmentation can produce significant gains in thrust. With an afterburner operating as described in this Appendix, a 33% increase in thrust to 36 lb for the J1 Engine and a 43% increase to 96 lb for the J2 Engine is predicted.

APPENDIX C

Instrumentation System

Parameter Sensitivity Analysis

An analysis of the effect on inaccuracy in the measurement performance data was made. The percent change of the independent variables necessary to produce a 5% change in the dependent parameter was determined. The calculation was made using data from Carpet with the J2 Engine for $P_{rc} = 3.31$. Only one independent variable was allowed to change at a time in the results shown. The form of the thrust equation as used for this analysis is

$$F_n = (W_a + W_f) \sqrt{2 C_p T_{o5} \eta_n \left[1 - \left(\frac{1}{P_{rn}} \right)^{\frac{\gamma-1}{\gamma}} \right]} \quad (3)$$

and C_p are from the Thermodynamic Gas Tables (Ref 7). To change F_n by 5%, the independent variable W_a had to change by 5.1% or W_f had to change by 22%, etc.

$$\begin{aligned} w_a &= 5.1\% \\ w_f &= 22\% \\ T_{o5} &= 10\% \\ P_{rc} &= 4.5\% \end{aligned}$$

The same procedure was used for the equations on the following page. A low percent change in the independent variable indicates the variable will have to be measured with more accuracy.

$$SFC = \frac{W_f}{F_n} \quad (4)$$

$$\begin{aligned} W_f &= 5\% \\ F_n &= 5\% \end{aligned}$$

$$\eta_c = \frac{(P_{rc})^{\frac{\gamma-1}{\gamma}} - 1}{\frac{T_{o3}}{T_{o2}} - 1} \quad (5)$$

$$\begin{aligned} P_{rc} &= 5.3\% \\ \left. \begin{aligned} T_{o3} \\ T_{o2} \end{aligned} \right\} &= 1.8\% \end{aligned}$$

$$\eta_t = \frac{1 - \frac{T_{o5}}{T_{o4}}}{1 - \left(\frac{1}{P_{rt}} \right)^{\frac{\gamma-1}{\gamma}}} \quad (6)$$

$$\begin{aligned} P_{rt} &= P_{o4}/P_{o5} \\ \left. \begin{aligned} T_{o4} \\ T_{o5} \end{aligned} \right\} &= 0.6\% \\ P_{rt} &= 2.2\% \end{aligned}$$

$$\eta_b = \frac{C_p}{Q} \frac{W_a}{W_f} (T_{o4} - T_{o3}) \quad (7)$$

$$\left. \begin{aligned} C_p \\ W_a \\ Q \\ W_f \end{aligned} \right\} = 5\%$$

$$\begin{aligned} T_{o3} &= 8.3\% \\ T_{o4} &= 3.1\% \end{aligned}$$

$$\Delta P_{ob} = 1 - \frac{P_{o4}}{P_{o3}} \quad (8)$$

$$\begin{aligned} P_{o4} &= 5\% \\ P_{o3} &= 5\% \end{aligned}$$

The parameter sensitivity analysis indicates the independent variables for thrust, and specific fuel consumption will not have to be measured with high accuracy and will show little effect of instrumentation error. Thrust as calculated from the equation, has a greater possibility of instrumentation error build-up. This analysis shows that the independent variables in equation 5 and 6 show significantly low percent change. The parameters also have a large probability of error build-up due to the number of independent variables in the equations.

Instrumentation And Measurement Accuracy

Bellmouth (W_a). The J1 Engine bellmouth was designed as a long-radius flow nozzle as defined by the American Society of Mechanical Engineers (Ref 8). It was constructed of aluminum with a throat diameter of 2.625 inches. Static pressure in the throat was measured and related to mass flow by the calibration curve shown on Fig 30. For calibration, the bellmouth, mounted on the turbocharger compressor housing, was connected to a vacuum system with an orifice plate and operated at various mass flows.

A wooden bellmouth, 4.057 in. throat diameter, was used for the J2 Engine. The calibration correction factor from the J1 Engine bellmouth was also used for this bellmouth with the resulting curve shown on Fig 31. Accuracy of the mass flow data is estimated to be within 2%.

Fuel Meter And Nozzle. The primary means of measuring the fuel flow was with a flow meter in the fuel line. This meter was calibrated by spraying fuel into a calibrated container and recording the time with a stop watch. The calibration curve shown as Fig 32 indicates considerable data scatter. The points on this chart indicate a possible band width of 21% at the low end and 4% at the high flow rates.

The fuel flow data is estimated to be within 5% at the points where the data was taken during the test.

Nozzle response to pressure for the nozzles used in the test is shown on Fig 33. The lower point on each plot is the pressure where the nozzle spray pattern begins to neck down

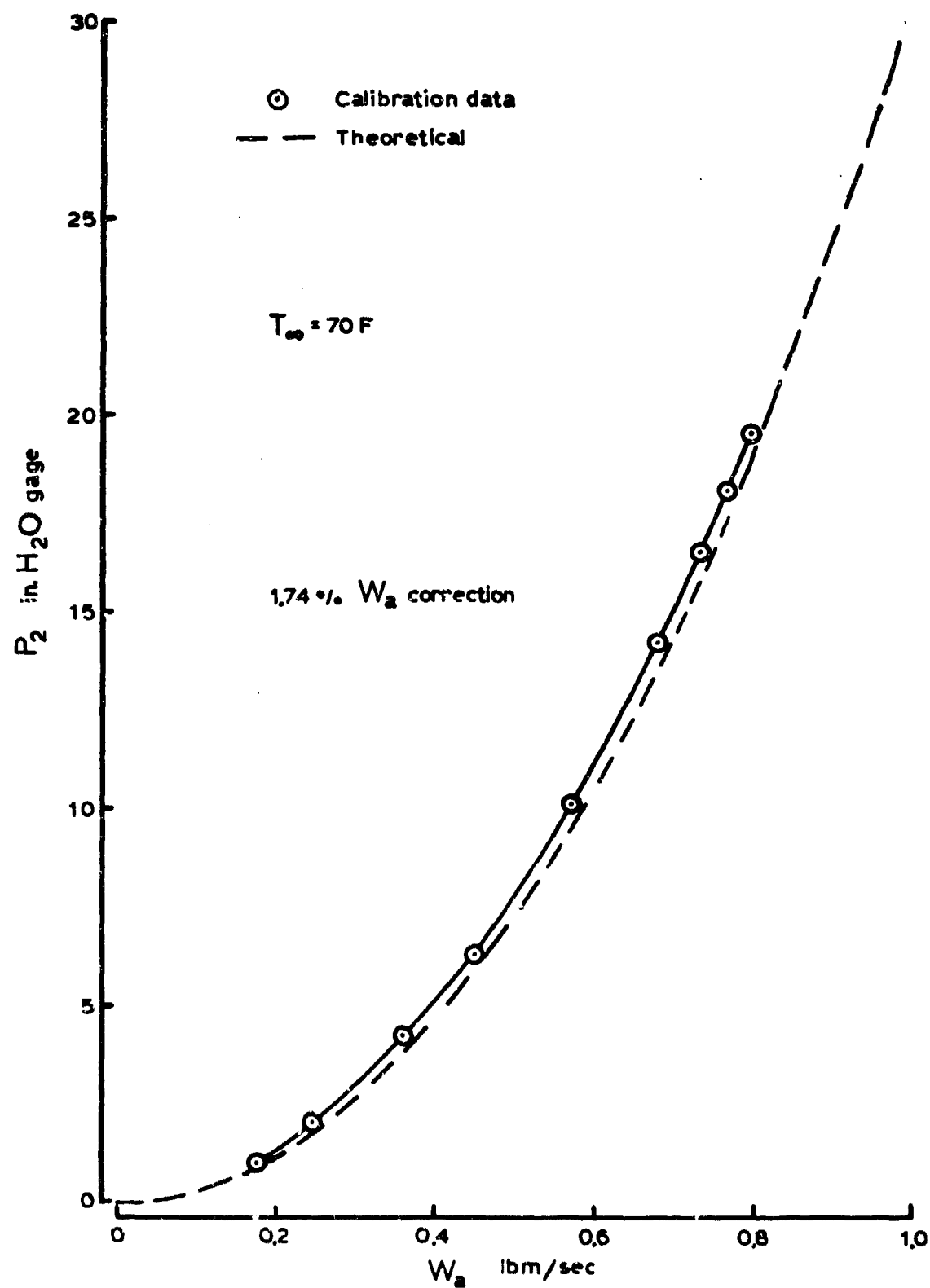


Fig 30: J1 Engine Bellmouth Characteristic Curve

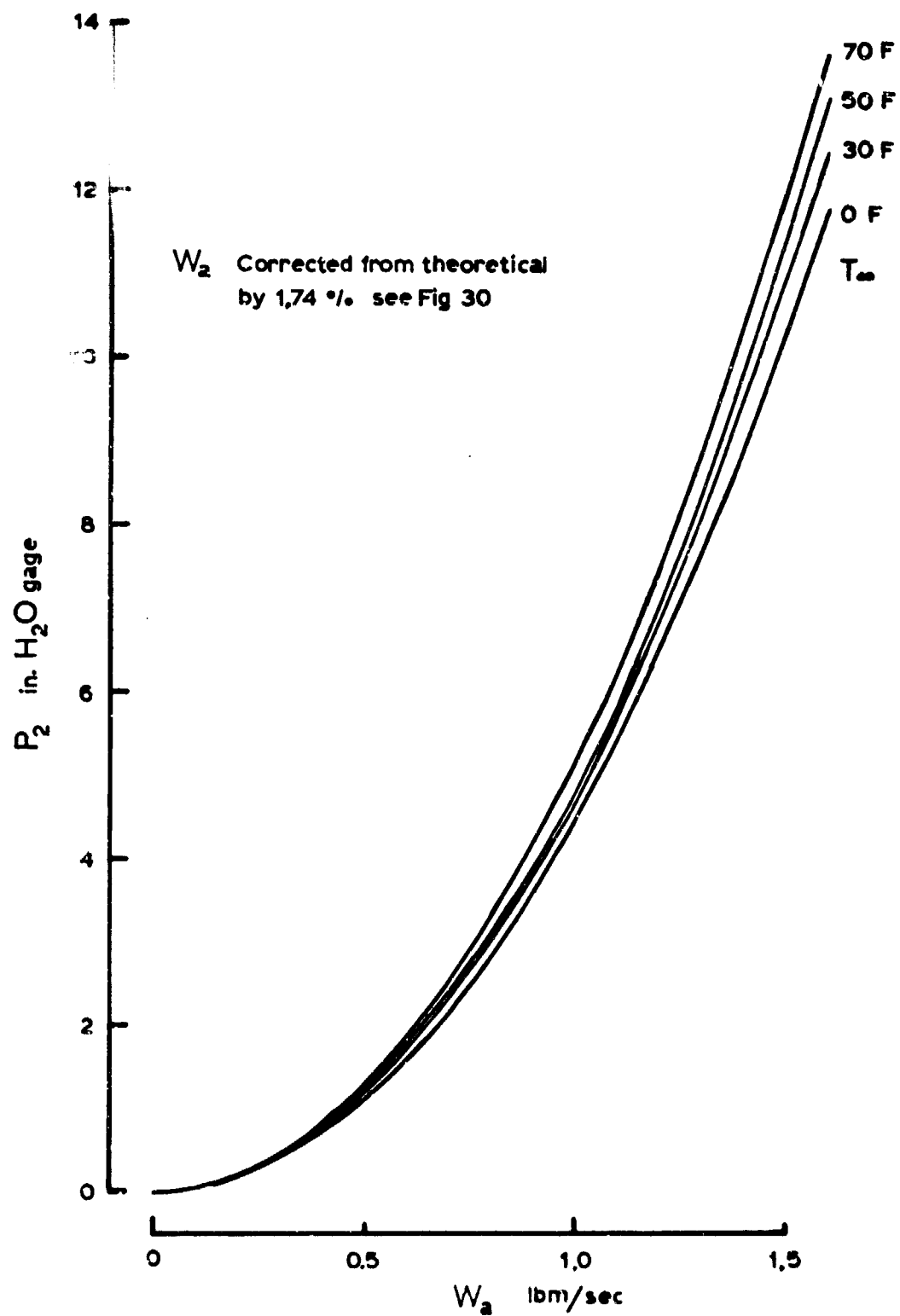


Fig 31: J2 Engine Bellmouth Characteristic Curve

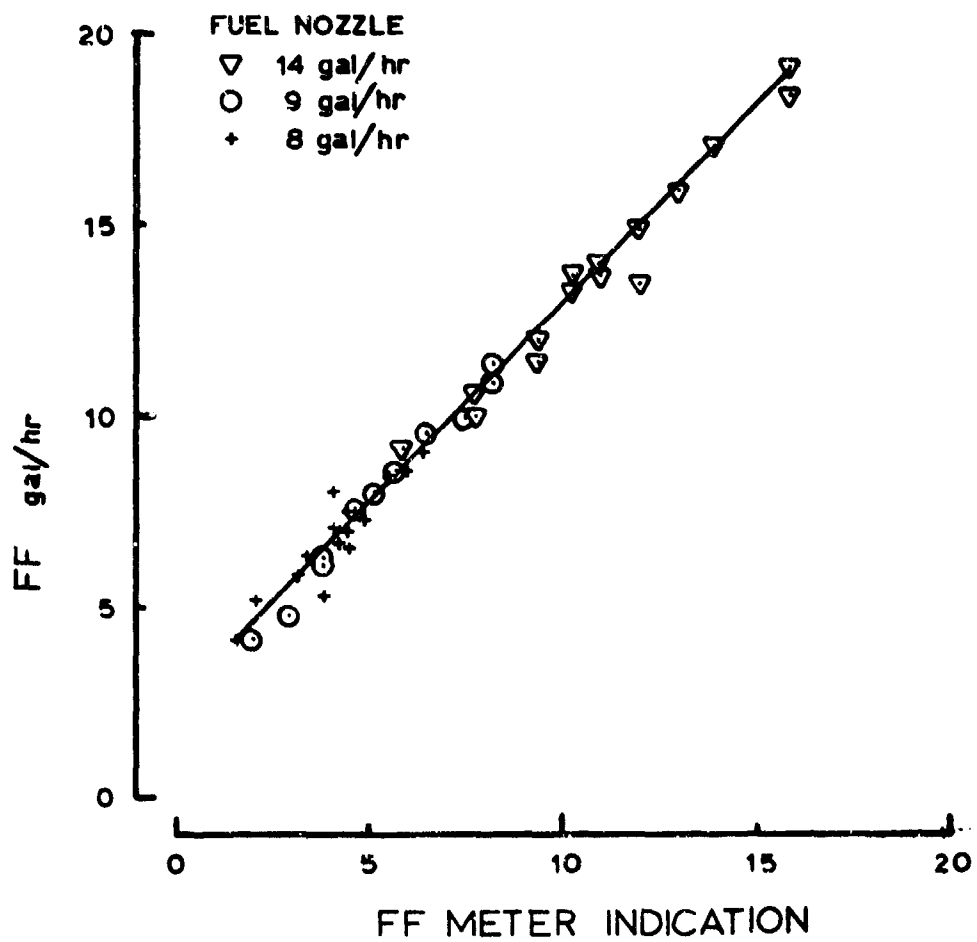


Fig 32: Fuel Flow Meter Calibration Curve

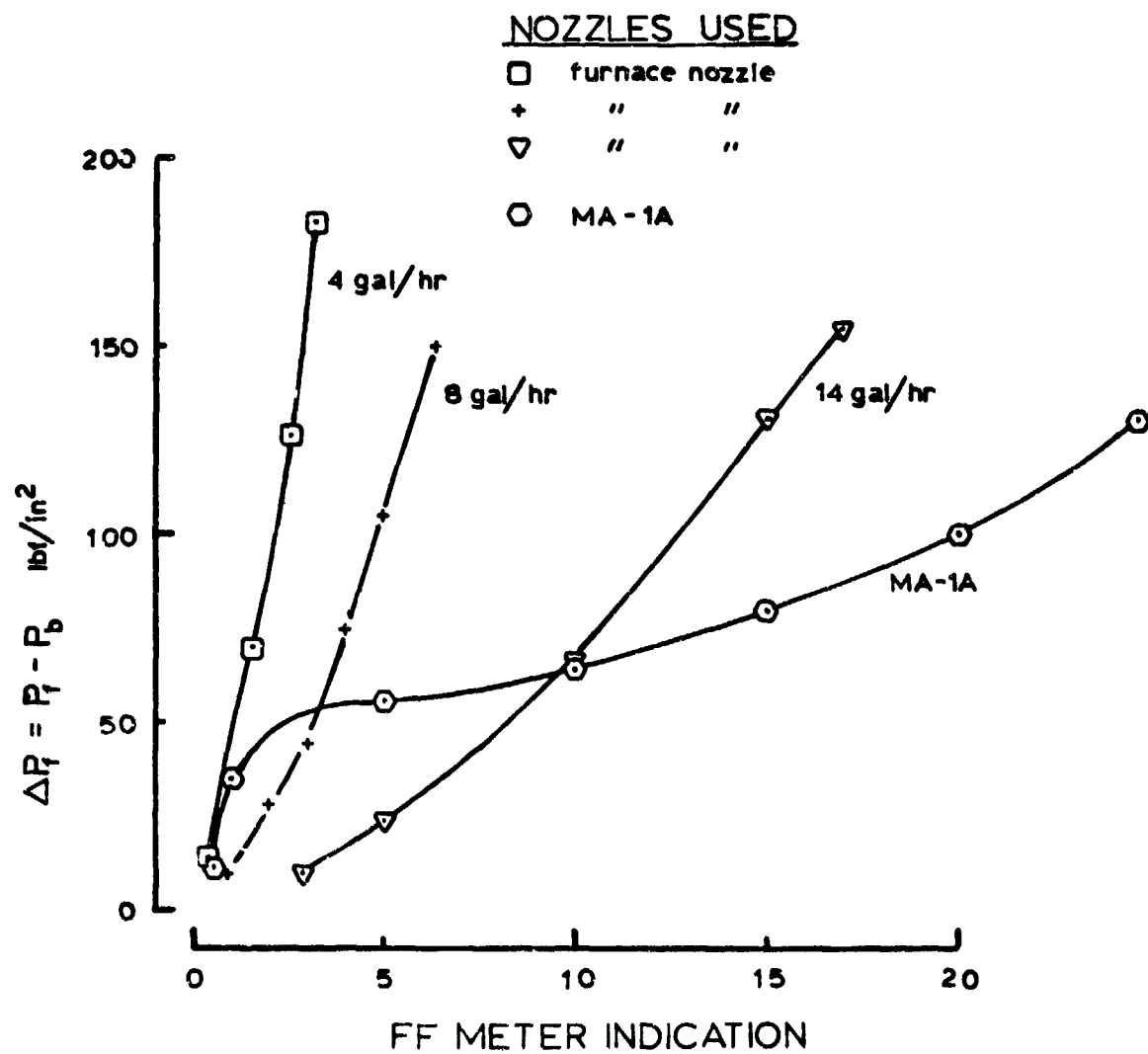


Fig 33: Fuel Nozzle Performance

and the nozzle becomes ineffective. The MA-1A fuel nozzle, as compared to the furnace nozzles, holds its spray pattern to a lower pressure. Also, it gives a larger spray angle and a much smaller droplet size.

Pressure. Fig 8 gives the approximate locations for the pressure taps used to collect performance data. All the pressure taps installed in the engine equipment were 1/16 ID holes with a 1/8 in. stainless steel tube used to connect the hole to 1/4 in. plastic tubing. The stainless steel tubes were necessary to isolate the plastic tubing from high temperature engine parts. Dial gages were used to read the combustion chamber plenum pressure, the static pressure aft of the turbine, static pressure in the transition section between the combustion chamber and the turbine, and fuel pressure. The bellmouth used 1/16 ID static pressure taps connected with 1/4" plastic tubing to a water manometer. The accuracy to which the dial gages were read, 1 in. Hg or 1 psi, was used in the error analysis. This gives a possible 3% error for P_B and P_5 ; P_4 and P_{fn} have a possibility of 5% error, with a possibility of 2% error for P_f . From these results, possible error of the pressure ratio terms are $P_{rc} = 1\%$, $P_{rn} = 2\%$ and $\Delta P_i = 8\%$.

Temperature. Temperature was measured using shielded Chromel-Alumel thermocouples for the hot flow and an open wire Iron-Constantan thermocouple at the compressor exit. These probes were positioned at the peak temperature point in the duct. No attempt was made to obtain an average temperature for the gas. These thermocouples were connected to Honeywell recorders to plot the temperature data. The recorders were calibrated before each run with a voltage source. Checks made with the Iron-Constantan thermocouple and boiling water indicated the measurements were within the accuracy given in Ref 9.

However, the recorders could not be read closer than approximately 5°F for the low temperature Iron-Constantan, and approximately 20°F for the high temperature Chromel-Alumel. Also, instabilities in the gas flow and small changes in the fuel flow delivered to the engines caused errors in temperature readings for the data points. The inlet temperature was measured with a mercury thermometer. Low temperature measurements are estimated to be within 2%, high temperature measurements within 5% for the combustion chamber exit, and within 3% for the nozzle exit where the gas flow was more stable. These estimates apply for the engine in stable operating conditions. When the machine was unstable, the low temperatures are estimated to be within a band of 50 °F for the data given, the high temperatures within a 200 °F band.

Direct Thrust Measurement. The thrust measurement device was a simple cantilever beam with a strain gage attached. The lower end of the beam was attached to the thrust stand structure. Part of the engine frame was extended forward of the bellmouth so the upper end of the beam was aligned with the engine centerline. The engine frame was hung from the thrust stand with levers attached to ball bearings allowing free movement.

The strain gage was fed through an amplifier and bridge network to a recorder that indicated the results on a strip of light sensitive paper. With the paper running at a constant rate, thrust vs. time was plotted. The thrust stand was calibrated by hanging weights from a cable connected to the engine

frame. The calibration plots were made just after the run, so all flexible lines and tubes were still at operating temperature and pressurized. These measurements were within 4% at the 30 lb and 2.5% at 50 and 60 lb of thrust.

Calculated Thrust Measurement. Thrust was calculated using Eq 14, see Appendix D. Using performance data from the Carpet deck for the J2 Engine at $P_{rc} = 3.31$, the equation was checked with the maximum variation for each variable to give a maximum thrust error of 7%. The difference in T_6 and T_{06} at $M = 0.49$ (see Table III) was used for the maximum possible difference between the peak temperature measured and the average temp of the gas. Also, $T_{05} = T_{06}$, so the temperature was measured at station 6 (see Fig 8) to reduce the effect of the nonuniform flow field.

APPENDIX D

Detailed Calculations And Data Reduction

Mach Number Calculation (Table III)

The Mach number (M) at various locations in the turbocharger engine ducts was calculated using the Weight Flow Parameter $\frac{W_a \sqrt{T_0}}{P_0 A} R$ from Ref 10 with W_a , T_0 , and P_0 taken from Carpet data. The areas in Table III were calculated from measurements of duct diameters in the engines and used in the parameter. "R" was considered a constant for these calculations.

Data Reduction Equations

The data reduction equations on the following page were used to reduce the data listed as Table V in Appendix A.

Data Reduction Equations

$$\delta = P_2 / 29.92 \quad (9)$$

$$\theta = T_{\infty} / 519 \quad (10)$$

$$W_a = \text{Function of } P_2 \text{ and } T_{\infty} \quad (\text{see Fig 30})$$

$$W_{\text{accr}} = \frac{W_a \sqrt{\theta}}{\delta} \quad (11)$$

$$P_{rc} = \frac{P_3 + P_{\infty}}{P_{\infty}} \quad (12)$$

$$P_{rn} = \frac{P_5 + P_{\infty}}{P_{\infty}} \quad (13)$$

$$\frac{F_n}{W_a} = (1+f) \sqrt{2 C_p T_{05} \eta_n \left[1 - \left(\frac{1}{P_{rn}} \right)^{\frac{\gamma-1}{\gamma}} \right]} \quad (1)$$

$$f = .02$$

$$C_p \text{ and } \gamma \text{ from Thermodynamic Gas Tables @ } T_{05}$$

$$T_{05} = T_{06} = T_6$$

$$\eta_n = 1.0$$

$$\frac{F_n}{W_a} = 7.093 \sqrt{C_p T_{05} \left[1 - \left(\frac{1}{P_{rn}} \right)^{\frac{\gamma-1}{\gamma}} \right]} \quad (14)$$

$$F_n = \left(\frac{F_n}{W_a} \right) W_a \quad (15)$$

$$F_{ncor} = \frac{F_n}{\delta} \quad (16)$$

$$SFC = \frac{FF (6.6)}{F_n} \quad 6.6 = \text{specific weight of JP-4 @ 30 F}$$

$$FF = \text{Fuel flow rate (gal/hr) = function of flow meter} \quad \text{see Fig 32}$$

$$W_{fcor} = W_f / \delta \sqrt{\theta} \quad (17)$$

$$SFC = \frac{W_{fcor}}{F_{ncor}} \quad (18)$$

VITA

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PII Redacted

[REDACTED] He completed grade school and high school in Delhi, Oklahoma and graduated from Oklahoma State University with a Bachelor of Science in Mechanical Engineering. After graduation, he worked for the Boeing Company in Seattle, for three years as an Aeronautical Engineer. He then joined the Air Force and after completion of pilot training in 1969, served as a KC-135 Aircraft Commander at Fairchild AFB, and in Southeast Asia. He entered the Air Force Institute of Technology in August, 1974.

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